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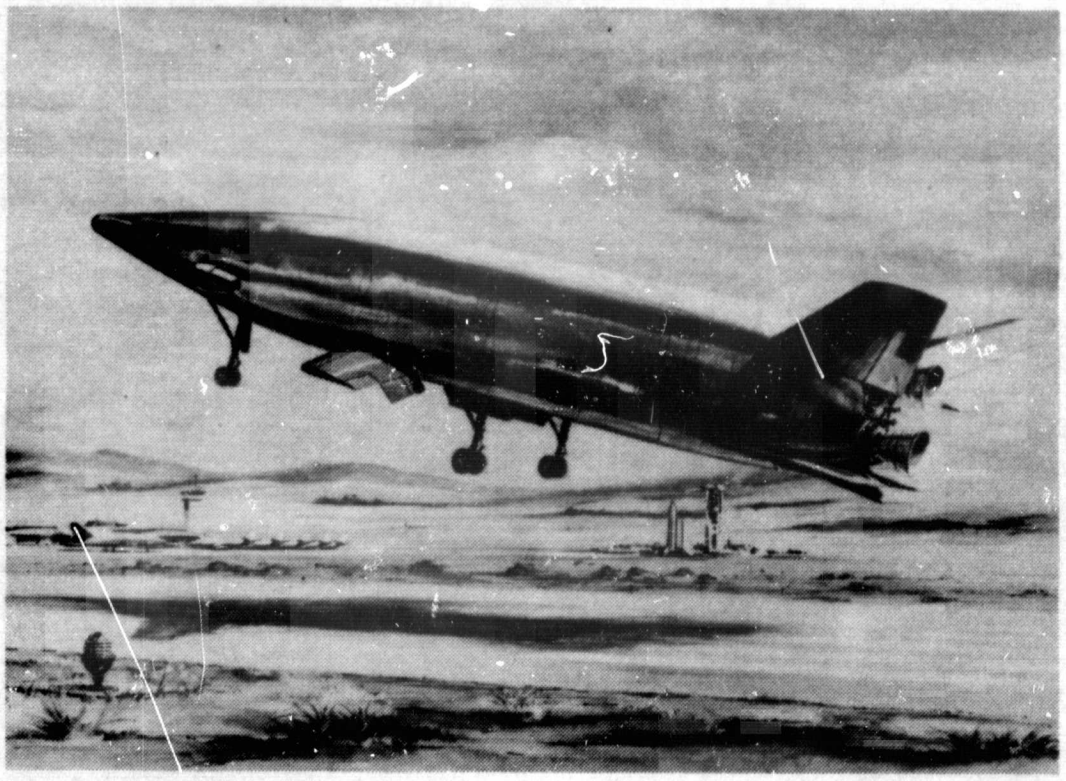
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SPACE SHUTTLE
FINAL TECHNICAL REPORT

VOLUME VI • PROPULSION ANALYSIS AND TRADEOFFS

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SPACE SHUTTLE
FINAL TECHNICAL REPORT

VOLUME VI ✦ PROPULSION ANALYSIS AND TRADEOFFS

31 October 1969

Prepared by
CONVAIR DIVISION OF GENERAL DYNAMICS
San Diego, California

FOREWORD

This volume of Convair Report No. GDC-DCB 69-046 constitutes a portion of the final report for the "Study of Integral Launch and Reentry Vehicles." The study was conducted by Convair, a division of General Dynamics Corporation, for National Aeronautics and Space Administration George C. Marshall Space Flight Center under Contract NAS 9-9207 Modification 2.

The final report is published in ten volumes:

Volume I	Condensed Summary
Volume II	Final Vehicle Configurations
Volume III	Initial Vehicle Spectrum and Parametric Excursions
Volume IV	Technical Analysis and Performance
Volume V	Subsystems and Weight Analysis
Volume VI	Propulsion Analysis and Tradeoffs (plus Confidential Addendum)
Volume VII	Integrated Electronics
Volume VIII	Mission/Payload and Safety/Abort Analyses
Volume IX	Ground Turnaround Operations and Facility Requirements
Volume X	Program Development, Cost Analysis, and Technology Requirements

Convair gratefully acknowledges the cooperation of the many agencies and companies that provided technical assistance during this study:

NASA-MSFC	Aerojet-General Corporation
NASA-MSC	Rocketdyne
NASA-ERC	Pratt and Whitney
NASA-LaRC	Pan American World Airways

The study was managed and supervised by Glenn Karel, Study Manager, C. P. Plummer, Principal Configuration Designer, and Carl E. Crone, Principal Program Analyst (all of Convair) under the direction of Charles M. Akridge and Alfred J. Finzel, NASA study co-managers.

ABSTRACT

A study was made to obtain a conceptual definition of reusable space shuttle systems having multimission capability. The systems as defined can deliver 50,000-pound payloads having a diameter of 15 feet and a length of 60 feet to a 55-degree inclined orbit at an altitude of 270 n.mi. The following types of missions can be accommodated by the space shuttle system: logistics; propellant delivery; propulsive stage delivery; satellite delivery, retrieval, and maintenance; short-duration missions, and rescue missions.

Two types of reusable space shuttle systems were defined: a two-element system consisting of a boost and an orbital element and a three-element system consisting of two boost elements and an orbital element. The vehicles lift off vertically using high pressure oxygen/hydrogen rocket engines, land horizontally on conventional runways, and are fully reusable. The boost elements, after staging, perform an aerodynamic entry and fly back to the launch site using conventional airbreathing engines. Radiative thermal protection systems were defined to provide for reusability. Development programs, technology programs, schedules, and costs have been defined for planning purposes.

During the study, special emphasis was given to the following areas: System Development Approaches, Ground Turnaround Operations, Mission Interfaces and Cargo Accommodations/Handling, Propulsion System Parameters, and Integrated Electronics Systems.

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SUMMARY

Design, analysis, and tradeoff studies have been made to determine the optimum characteristics of the propulsion subsystems for two-element and three-element space shuttle vehicles. All of these studies are documented in this volume. The most significant results of these studies are as follows:

a. Main Propulsion Tradeoffs Using High Chamber Pressure Bell Engines.

1. Thrust/Weight Ratio. Thrust/Weight (F/W) ratio tradeoff studies were made assuming both fixed-thrust engines, where thrust/weight was varied by changing the number of engines, and rubber engines, where thrust of a constant number of engines was allowed to vary. For all vehicles studied, minimum gross liftoff weight (GLOW) was obtained at a launch F/W of approximately 1.5, but minimum vehicle dry weight, a better cost indicator, was obtained at approximately 1.35. For the final two-stage vehicles, minimum GLOW was obtained with 16 engines, but minimum dry weight was obtained with 15 engines, the selected configuration.

2. Area Ratio. Area ratio trades were made for both three-element parallel burn FR-1 vehicles and two-stage sequential burn FR-3 vehicles. For the FR-1 vehicles, use of identical two-position nozzles on all three elements was found to give near-optimum performance and lowest cost. For the FR-3 two-stage vehicles, a compromise nozzle giving a low area ratio of 60 to 100 for the booster and 200 for the orbiter was found to give near-maximum performance and lowest cost. The same primary nozzle is used for both orbiter and booster, but different extensions are used. Selection between a fixed or movable extension for the booster was found to require more detailed study.

3. Mixture Ratio. A mixture ratio of 7.0 in all elements was found to give minimum launch weights for both FR-1 and FR-3 vehicles. A mixture ratio of 6.5 was selected for the final vehicles, since it is within the existing technology range.

4. Liftoff Rating. Engines designed to operate over a mixture ratio range at some specified nominal thrust have a peak thrust capability without reduction of engine life at an intermediate MR. Use of this peak thrust capability at liftoff and early in booster flight can

improve vehicle performance compared to use of the nominal thrust. At fixed payload, launch weight is reduced by as much as 4.5 percent for LR-129 design characteristics (design MR range of 5.0 to 7.0). For a narrower design MR range, a smaller but significant gain should be obtained.

5. Net Positive Suction Head (NPSH). There is no incentive to reduce the NPSH requirements below the current requirement. There is a potential for improved performance through reduction in inducer and inlet duct weights at higher design NPSH requirements.

6. Engine Thrust. The use of engines of specified fixed thrusts of 300,000 lb, 400,000 lb, and 500,000 lb was studied for FR-1 and FR-3 vehicles. There were no significant identifiable differences beyond "unit engine" effects in performance or cost between these engines and engines in the 600,000-lb to 1,000,000-lb thrust class. There were some advantages in using the smaller engines for FR-3 since they provide more flexibility in proportioning thrust between booster and orbiter. The 500,000-lb thrust engine was recommended on the basis of reduced probability of catastrophic failure over the life of the space shuttle system.

7. Use of Crossfeed. Use of crossfeed vs no crossfeed was evaluated on three-element FR-1 and FR-4 vehicles. Use of crossfeed and parallel burn of all engines on propellants supplied from the booster tanks was found to reduce launch weight of common-element vehicles (same number of engines in booster and orbiter elements) by 6-9 percent or more. For vehicles with engine arrangements optimized for crossfeed or no crossfeed, differences in performance are minor, the exact differences depending on detail weights added for crossfeed plumbing and associated hardware vs added engine weight for no crossfeed. Specific vehicles analyzed were about 1.5 percent lighter without crossfeed.

8. Orbit Maneuver Propulsion. Studies were made to determine the most efficient method for using the main engines to perform the orbit maneuvers required for the basic space shuttle mission. Approaches considered were 1) combined use of the attitude control propulsion subsystem and a single main engine in a low-thrust pump-fed mode, 2) main engine in a pressure-fed idle mode, and 3) main engine in a combined pressure-fed/pump-fed low-thrust mode. Use of the combined pressure-fed/pump-fed approach was

found to give best performance, but the pressure-fed idle mode was identified as a significant development risk.

9. Use of Aerospike Engines. Use of single and dual-combustor aerospike engines at 2000 and 2500 psi chamber pressures was investigated and compared with the use of high P_c bell-nozzle engines for FR-1 and FR-4 vehicles. The integrated dual-combustor aerospike was found to be comparable to the bell at 2000 psi and superior at 2500 psi P_c if performance losses associated with secondary injection (which were not evaluated) were not considered. The button aerospike, designed to operate within the bell-engine installation envelope, was found to be inferior to the bell at 2000 psi, reducing payload capability obtainable with the bell by more than 35 percent. At a P_c of 2500 psi, payload reduction was 16 percent.

b. Attitude Control Propulsion Subsystem (ACPS) Studies. Vehicle, mission, control, and redundancy requirements were studied so that arrangements, size, and duty cycle of ACPS thrusters could be identified. To meet all of the requirements identified, 48 thrusters in the 2000 to 3000-lb-thrust class were required. Optimum chamber pressure for the thrusters was found to be in the 300 to 500-psi range. Use of gaseous O_2 and H_2 propellants was found to be desirable from the standpoints of performance, non-toxicity, low corrosiveness, and commonality with main propellants. A propellant supply system using a gas-phase compressor/accumulator combination, taking propellants from the main tanks, was found to be most applicable to space shuttle mission requirements.

The need for further study and technology work in the areas of vehicle control requirements, H_2/O_2 thrusters, and H_2/O_2 propellant feed system was identified.

c. Flyback and Landing Propulsion Studies. Requirements for flyback and landing engines for both booster and orbiter vehicles were studied. High-bypass ratio turbofan engines were found to be best suited to the booster flyback requirement on the basis of current and projected engine technology. For orbiter vehicles, short life, high-thrust-to-weight ratio engines of the lift-engine type were found to be most suitable. Only powered approach and landing capability (no go-around) is recommended for the orbiter vehicle. Use of hydrogen rather than JP-4 fuel was found to offer a potential 5 percent reduction in system launch weight. There is a need for technology work in adaptation of engines for boost and space environments, and in the use of hydrogen fuel.

SECTION 1

INTRODUCTION

During the Study of Integral Launch and Reentry Vehicles, design and tradeoff studies have been made of all aspects of propulsion for space shuttle vehicles. These studies have been made to support the normal vehicle design activity for the study and, at NASA direction, to develop and document propulsion tradeoff data, trends, and sensitivities in a number of areas. Significant results of all of these studies are documented in this volume. Confidential data is contained in an addendum to this volume.

Propulsion functions for the space shuttle may be conveniently divided into the following major areas:

a. Boost Propulsion. Boost propulsion provides all propulsion functions for the space shuttle from launch through injection at a 45 n. mi. perigee of an orbit with an apogee of 100 n. mi. For all space shuttle vehicles studied, the boost propulsion system has been based on use of high chamber pressure (high P_c) bell nozzle engines in booster and orbiter elements. Orbiter and booster engines have in all cases been identical except use of different nozzle expansion sections. For certain vehicles, the use of aerospike engines in both orbiter and booster elements has been investigated.

b. Orbit Maneuver Propulsion. Orbit maneuver propulsion is used for all on-orbit operations requiring significant velocity increments (ΔV). For the baseline mission, these include circularization in the 100 n. mi. phasing orbit, Hohman transfer firings from 100 n. mi. into 250 n. mi. circular orbits, and retrofire for entry from a 270 n. mi. orbit. All vehicles studied have used the main orbiter engine(s) operating in low thrust modes for these functions.

c. Attitude Control Propulsion. Attitude control propulsion subsystem (ACPS) provides orbiter attitude control during orbital and entry phases of flight, minor velocity increments for rendezvous and docking, drag makeup, propellant settling, and roll control during orbit maneuver firings. Use of both earth-storable propellants and O_2/H_2 systems has been studied. The current versions of all space shuttle orbiter vehicles use gaseous O_2/H_2 systems, with multiple thrusters clustered near the nose and tail of the vehicle.

d. Flyback and Landing Propulsion. All space shuttle booster vehicles studied fly back to the launch site after entry and land on conventional runways, using high bypass ratio turbofan engines mounted near the nose. Flyback ranges average 230 to 285 n. mi. All orbiter vehicles have engines to provide for powered landing, with go-around capability provided in some cases. Turbojet and high bypass ratio turbofan engines have been used in orbiter vehicles.

Study effort in main propulsion has been divided into the following major areas:

a. Tradeoff Studies. Design tradeoff studies have been made to aid in selection of optimum characteristics for high chamber pressure bell propulsion systems. These studies were made on FR-3 two-element vehicles without crossfeed and FR-1 three-element vehicles with crossfeed. Trade studies included thrust/weight ratios, area ratio, mixture ratio, nozzle type, and crossfeed.

b. Comparison of Aerospike and Bell Engines. Comparisons were made between integrated aerospike, "button" aerospike, and high P_c bell engines for three-element vehicles with crossfeed (FR-1) and without crossfeed (FR-4). Single- and dual-combustor aerospike engines at 2000 and 2500 psi chamber pressure were considered.

c. Effect of Engine Size. A study was made to examine the overall effects of use of bell engines of four different sizes in the three-element FR-1 vehicles and two-element FR-3 vehicles. Sea-level thrusts considered for FR-1 were 300K, 400K, 500K, and 984K pounds, and for FR-3 were 300K, 400K, 500K, and 602K pounds.

The main study effort in orbit maneuver propulsion has been to identify the most efficient method for use of the main engines in a low thrust mode. Approaches studied were:

a. Combined use of ACPS and main engine in a pump-fed 20 percent thrust mode.

b. Main engine in a pressure-fed idle mode.

c. Main engine in a combined pressure-fed/pump-fed idle mode.

The major study effort in attitude control propulsion has been to identify thruster sizes and arrangements meeting the imposed vehicle control requirements, to determine thruster duty cycles, and to define a gaseous O_2/H_2 propellant system satisfying the mission requirements.

The major study effort in airbreathing propulsion has been to identify requirements for flyback, go-around, and landing, to identify rubber engine characteristics which satisfy these requirements, and to select off-the-shelf engines available in the required time period which most nearly meet these requirements.

SECTION 2

PROPULSION DESIGN AND TRADE STUDIES

2.1 TRADE STUDIES

Design trade studies have been made to aid in establishing optimum characteristics for main and orbit maneuver propulsion for FR-1 three-element vehicles and FR-3 two-element vehicles. All of these trade studies have been made using high P_c bell nozzle engines. Engine performance data was based generally on LR-129 engine design characteristics obtained from Pratt and Whitney.

Trade studies documented in this section are:

	FR-1	FR-3
Thrust/Weight (Common Engines)	X	
Booster Thrust/Weight (Uncommon Engines)	X	
Orbiter Thrust/Weight (Uncommon Engines	X	
Booster Area Ratio, Two-Position Nozzles	X	X
Orbiter Area Ratio, Two-Position Nozzles	X	X
Booster Area Ratio, Fixed Nozzle	X	X
Orbiter Area Ratio, Fixed Nozzle	X	X
Mixture Ratio	X	X
Nozzle Contour	X	
Liftoff Overrating	X	
Orbit Maneuver Propulsion	X	
Crossfeed vs. No Crossfeed	X	
NPSH (Net Positive Suction Head)	X (FR-4)	

The FR-1 vehicles on which these trade studies were performed are shown in Figures 2-1 and 2-2. The two configurations are similar except for the use of six engines in a 2-2-2 booster-orbiter-booster arrangement in one and eight engines in a 3-2-3 arrangement in the other. In both vehicles, there is a high degree of commonality between booster and orbiter elements.

The FR-3 vehicle on which trade studies were made is shown in Figure 2-3. The booster uses 16 engines of 400,000-lb thrust at sea level and the orbiter uses three engines differing from booster engines only in the use of a different nozzle extension, giving a higher area ratio.

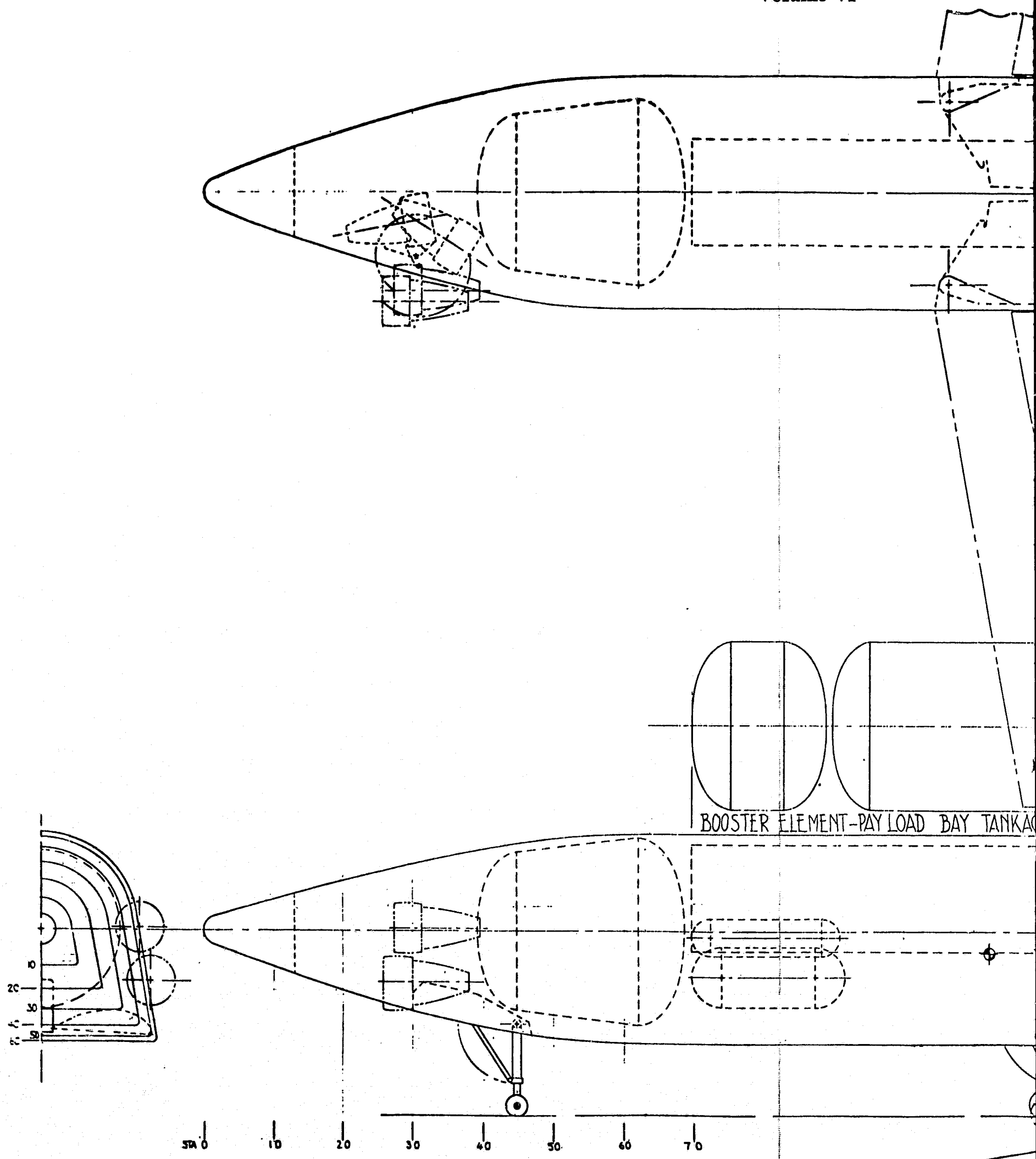
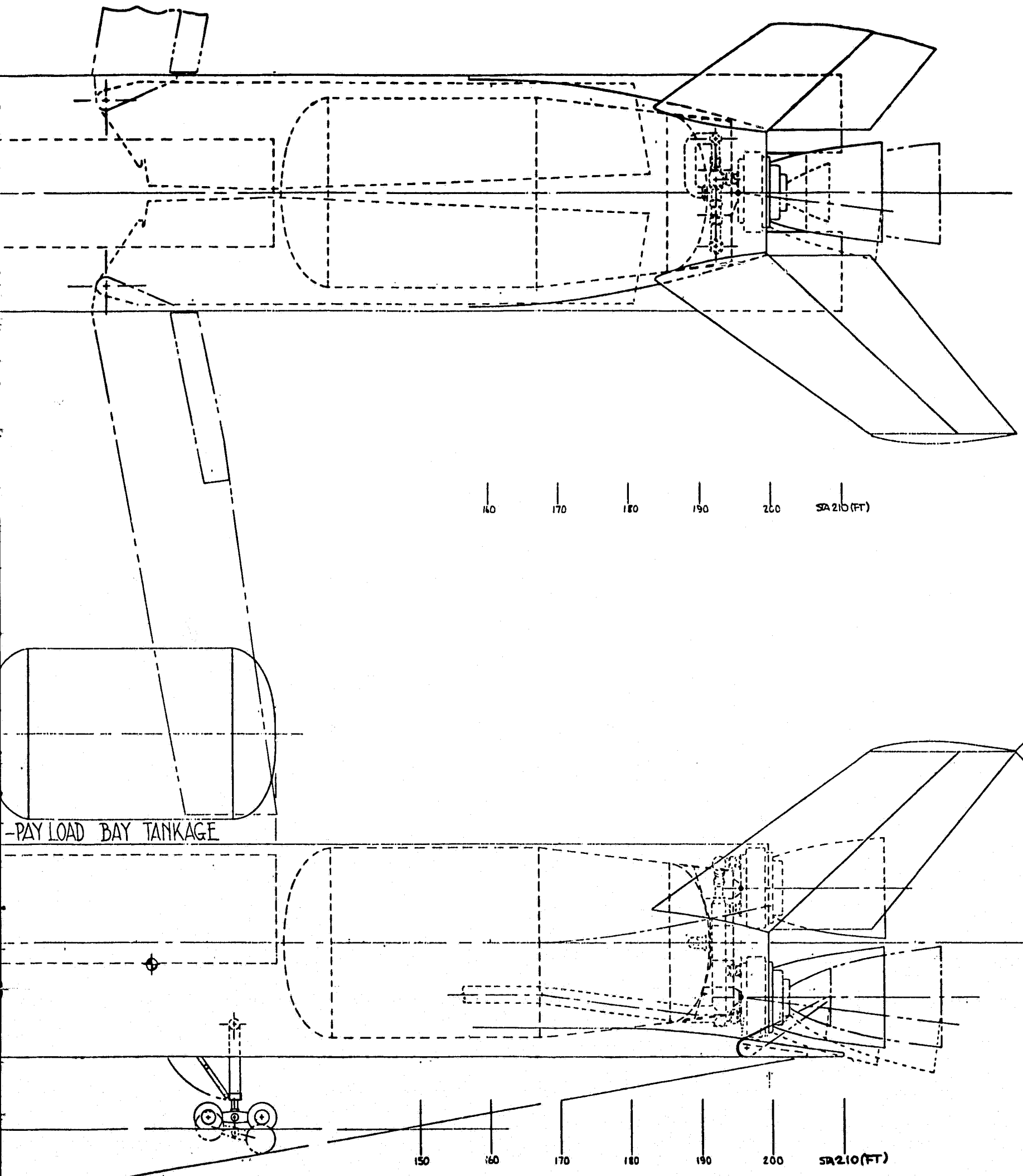


Figure 2-1. Six-Engine FR-1 Vehicle

FOLDOUT FRAME 1

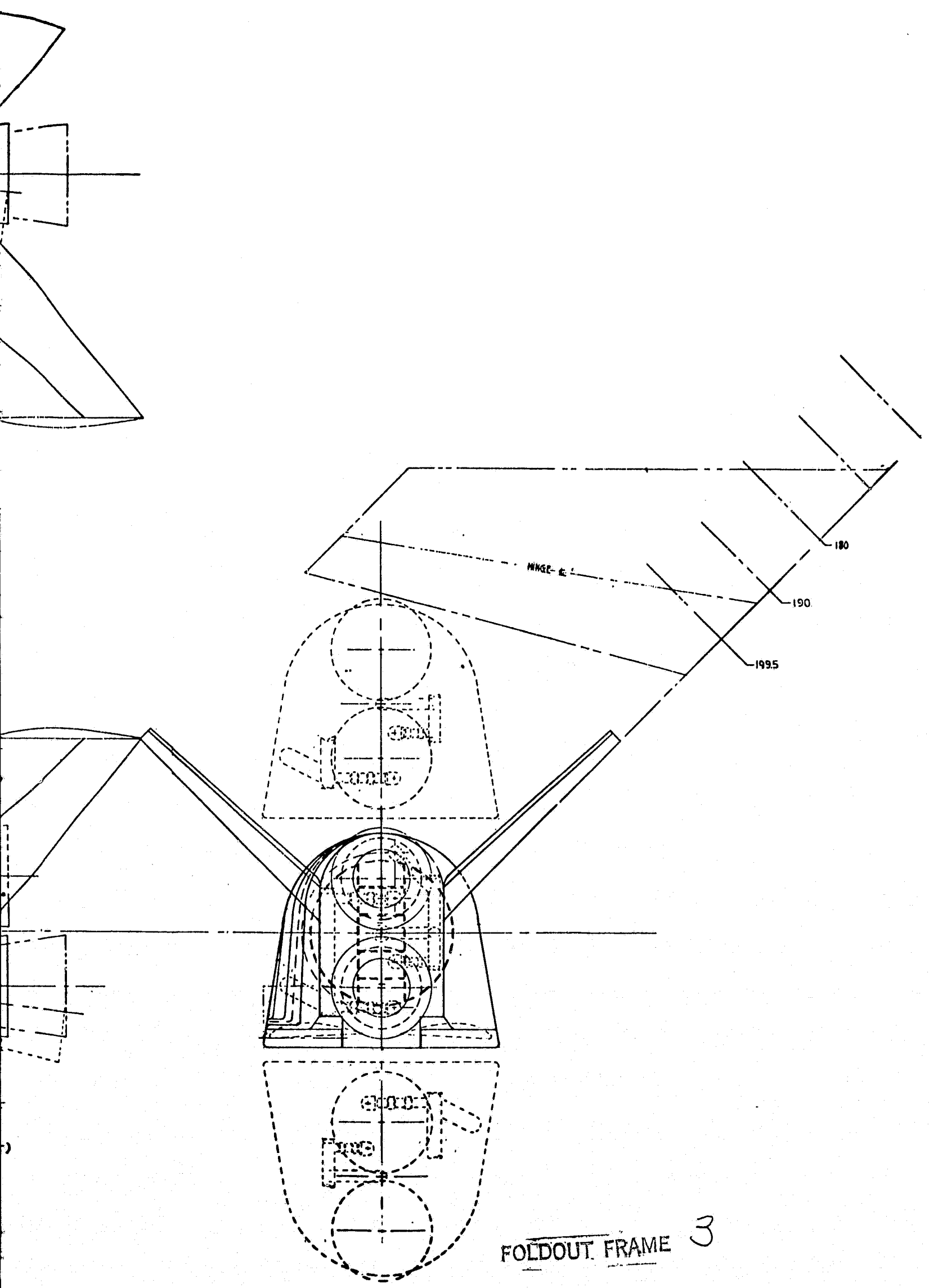
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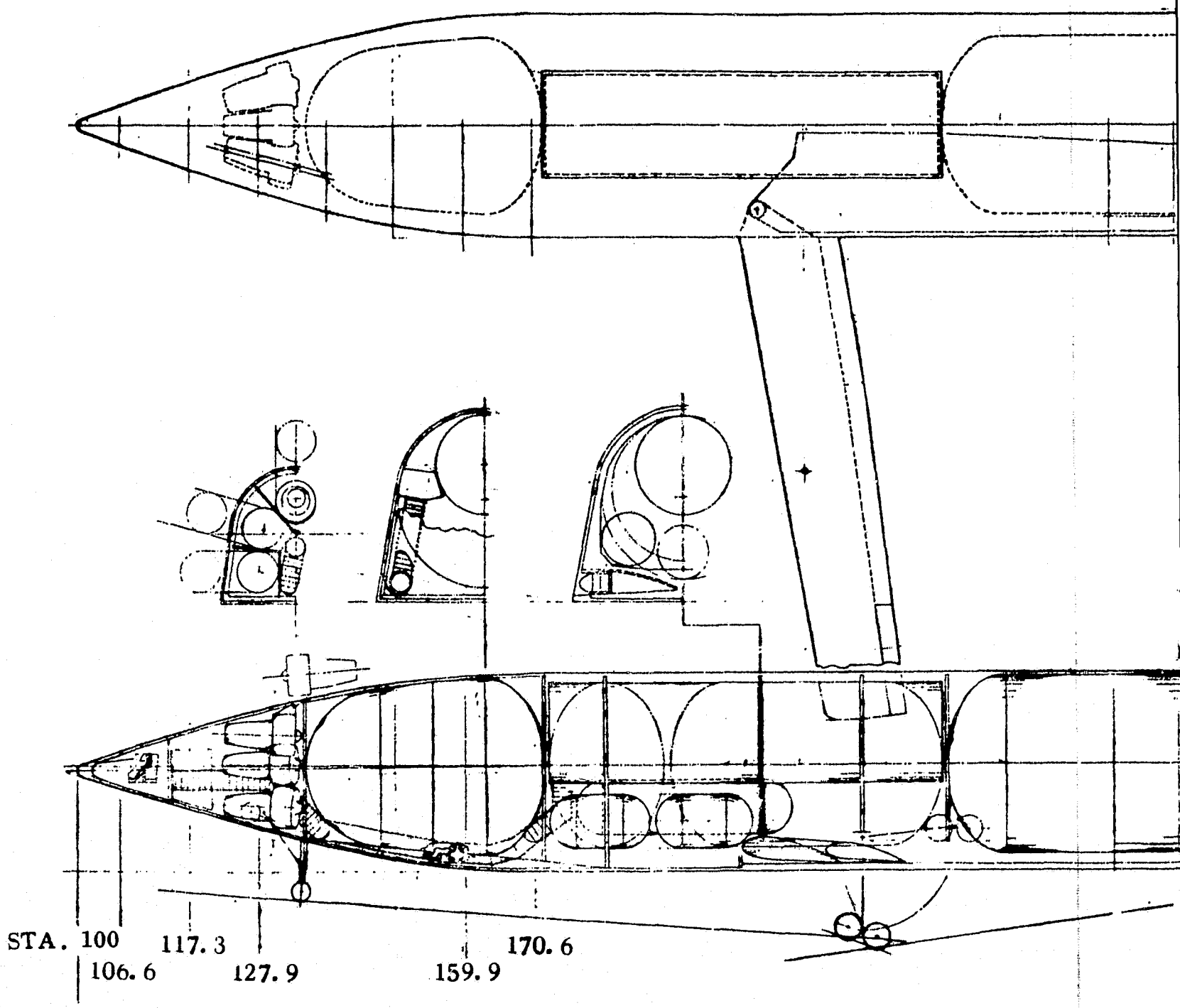
ORBITER ELEMENT

FOLDOUT FRAME 2

E

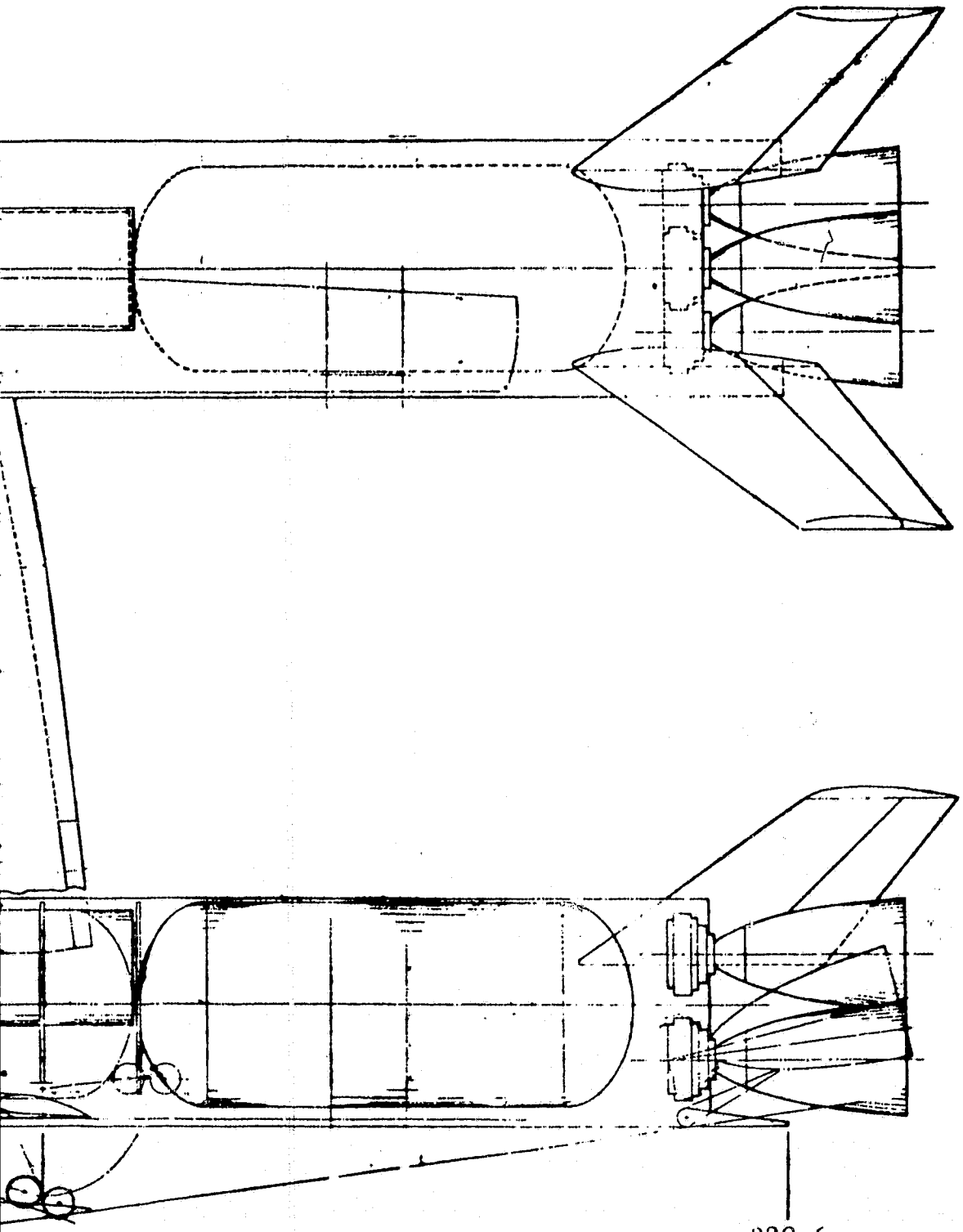


FOLDOUT FRAME 3



FOLDOUT FRAME 1

Figure 2-2. Eight-Engine



FOLDOUT FRAME 2

Figure 2-2. Eight-Engine FR-1 Vehicle

Figure 2-3. FR-3 Launch Configuration

2.1.1 THRUST/WEIGHT RATIO

2.1.1.1 FR-1 Parallel Burn Vehicles. Thrust/weight (F/W) tradeoff data has been generated for two different FR-1 parallel burn vehicles. Tradeoff factors considered were vehicle launch weight, vehicle inert weight, and engine thrust. When these parameters showed conflicting trends or different optimum values, sufficient cost information was developed to identify the driving parameter.

Vehicles on which the data was developed are the 8-engine 3 X 2 X 3 FR-1 (three engines on each booster, two engines on the orbiter) and the 6-engine 2 X 2 X 2 FR-1 (two engines on each element) shown in Figures 2-1 and 2-2. For the 3 X 2 X 3 configuration, both common engines (all engines identical) and uncommon engines (orbiter and booster engines of different thrust) were investigated. For the 2 X 2 X 2 configuration, only common engines were considered. For an FR-1 vehicle with equal-size elements and using identical engines in all elements, the orbiter F/W is uniquely determined by the relative number of engines in the orbiter and booster and the liftoff F/W ratio.

Figure 2-4 gives results of F/W optimization studies for both booster and orbiter with uncommon engines (i.e., engines of different thrust in orbiter and booster), where orbiter F/W can be uncoupled from liftoff F/W. These data show that the common engine configuration (points indicated on curve) gives very nearly optimum orbiter F/W ratios, so there would be no performance benefits for the development of two different engines.

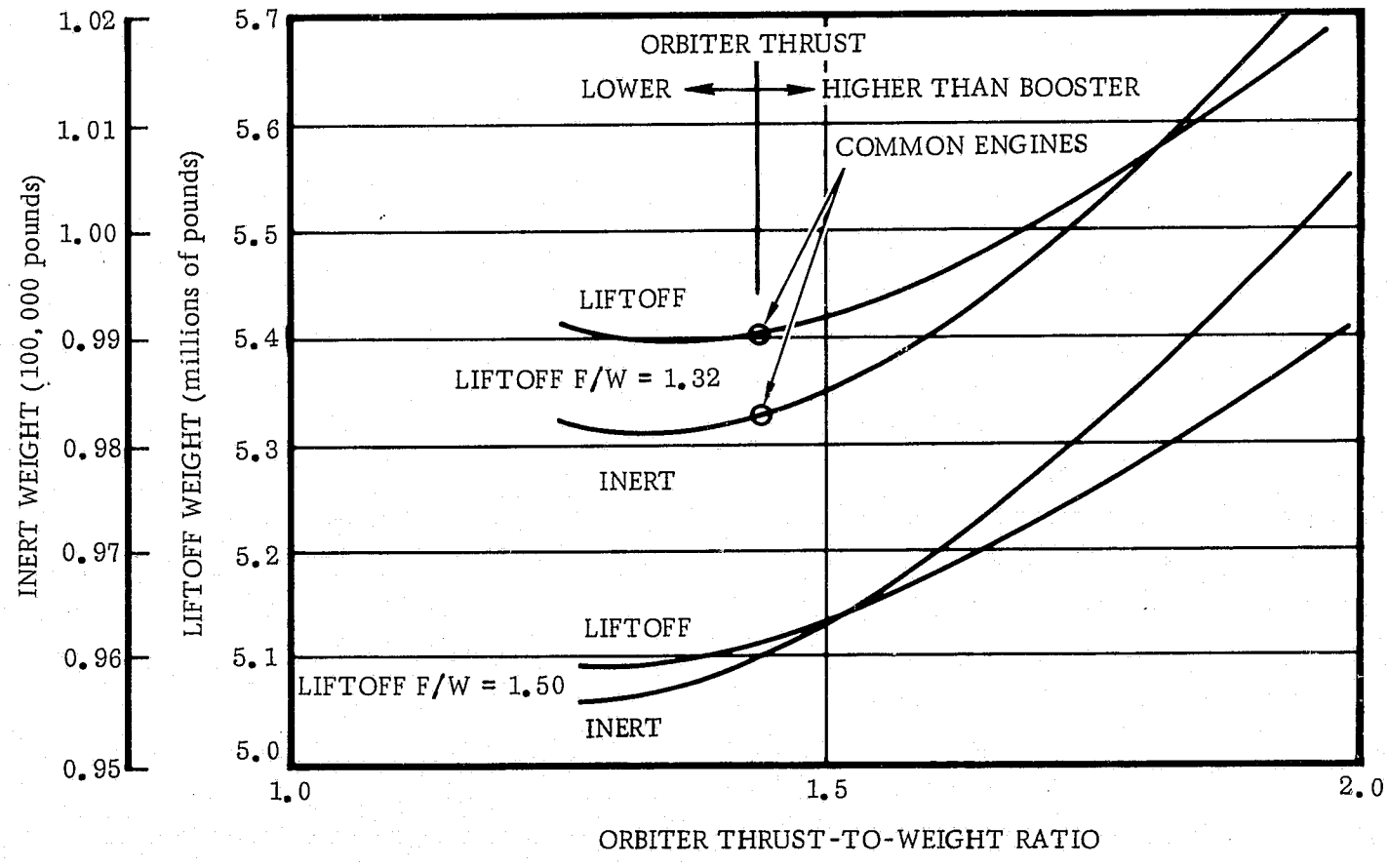


Figure 2-4. Effect of Orbiter F/W on Launch and Inert Weight (3 X 2 X 3 Configuration)

Figure 2-5 gives results of a liftoff F/W ratio optimization for the 2 x 2 x 2 configuration using common engines. The study was first performed assuming that an area ratio (ϵ) of 150 could be used on all vehicle elements. With this assumption, system launch weight was found to be a minimum at F/W = 1.48, where a shallow bucket in the launch weight versus F/W was found. The minimum launch weight at the 1.48 F/W was only four percent lower than at 1.25 (minimum value considered) and one percent lower than at 1.59, the maximum value considered. Vehicle inert weight was also found to be relatively insensitive to F/W ratio, with the minimum at F/W = 1.35.

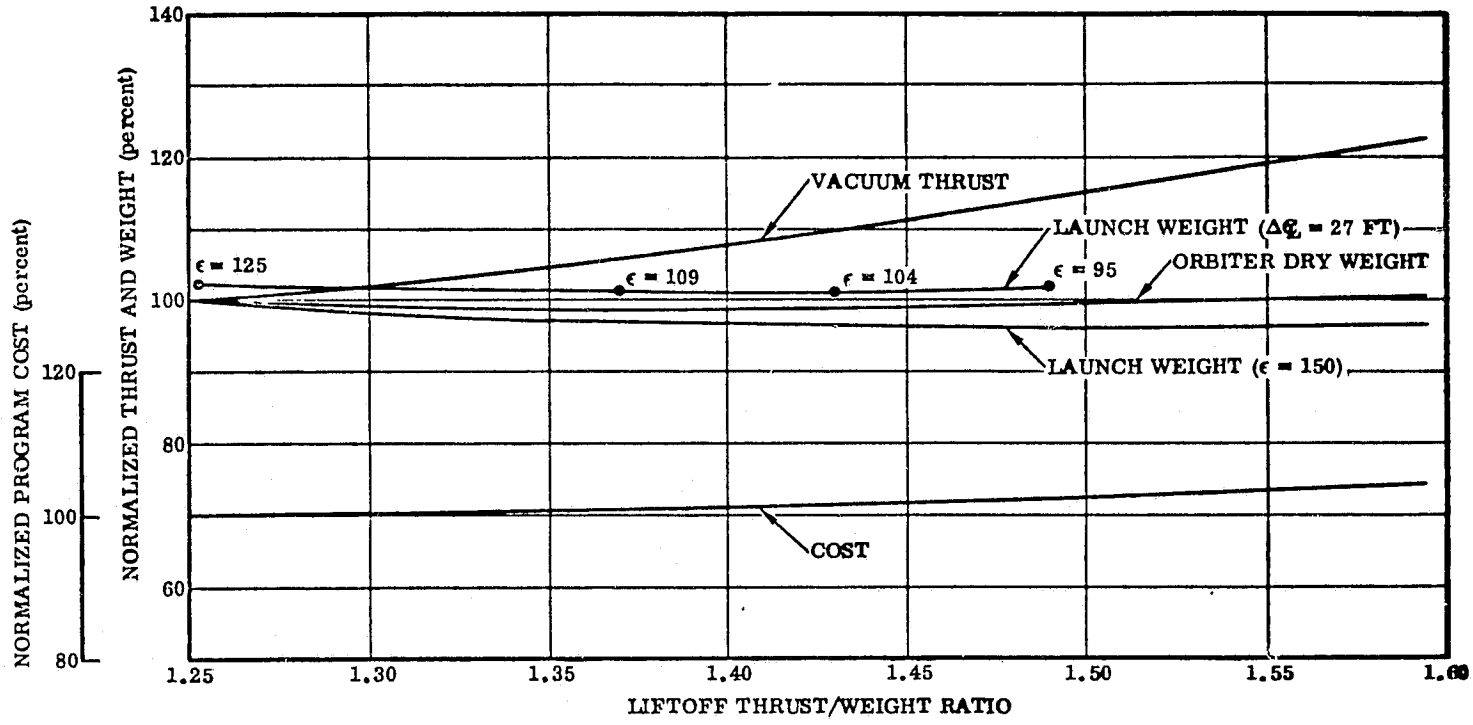


Figure 2-5. Effect of Liftoff Thrust/Weight Ratio on Weight, Thrust, and Program Cost, 2 x 2 x 2 Configuration

Engine thrust required, however, was almost a linear function of F/W and increased steadily by 22 percent over the 1.25 to 1.59 F/W range considered. A cost analysis showed that engine development cost was the driving parameter and minimum total program cost would be obtained with the lowest F/W considered. For the vehicle configuration studied, the assumed area ratio of 150 would not actually fit in the available space. The upper curve gives values of liftoff weight adjusted for area ratio effects, showing reduced sensitivity to F/W and minimum launch weight at a lower F/W ratio. Costs were not adjusted, but the effect would be to increase the slope of the curve. Balance of the vehicle is improved at the lower F/W ratios, since lighter engines can be used. It was concluded that the lowest F/W ratio commensurate with the engine-out requirements should be used for the 2 x 2 x 2 FR-1. For a minimum F/W of 1.16 with one engine out, the design liftoff F/W would be 1.392.

Figure 2-6 gives results of a liftoff F/W ratio optimization study for the 3 x 3 x 3 configuration using common engines. Results are similar to those for the 2 x 2 x 2

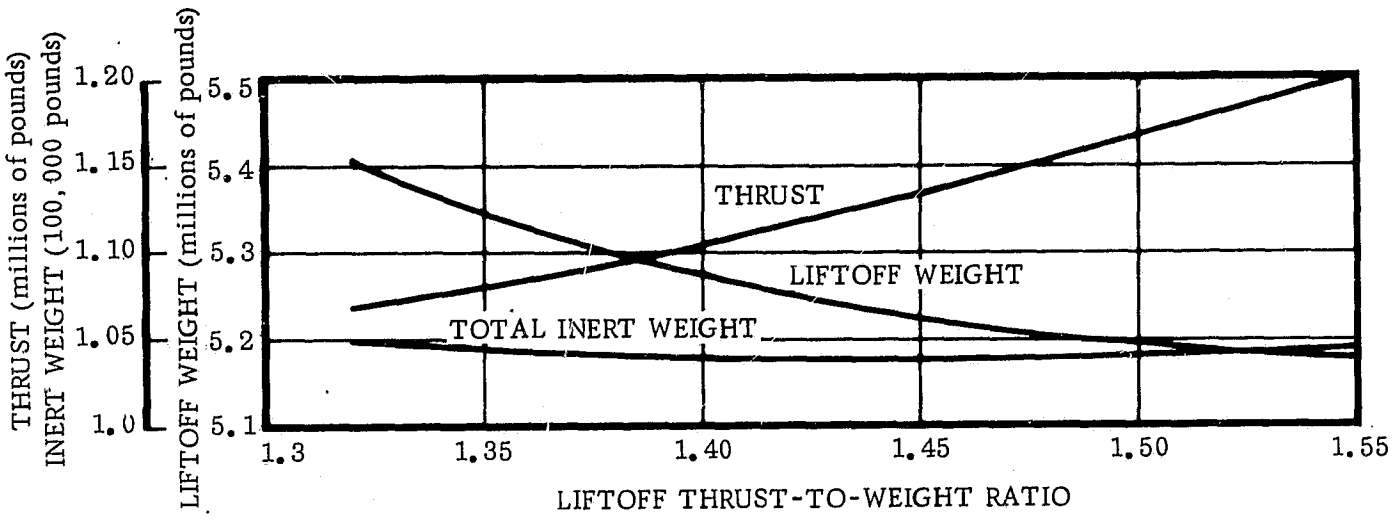


Figure 2-6. Effect of Liftoff F/W on Weight and Thrust (3 X 2 X 3 Configuration)

configuration, with a nearly constant inert weight and an engine thrust requirement decreasing with F/W over the range considered. It was again concluded that the minimum liftoff F/W consistent with engine-out requirements should be used; for F/W of 1.16 with one engine out, this is 1.324. With these selected liftoff F/W ratios, thrust required for the 6-engine 2 X 2 X 2 vehicle is 1,100,000 lb sea level (1,300,000 lb vacuum) and for the 8-engine 3 X 2 X 3 vehicle is 810,000 lb sea level (945,000 lb vacuum).

2.1.1.2 FR-3 Sequential-Burn Vehicles. For the final FR-3 two-stage sequential-burn vehicle, engines of fixed thrust were used, so the liftoff thrust-to-weight ratio was a function of the number of booster engines used. Figure 2-7 shows launch weight and total (sum of booster and orbiter) inert weight for 14, 15, and 16-engine boosters. Also shown is the launch weight vs. F/W for a 15-engine booster using "rubber" engines (thrust allowed to vary), developed to locate the optimum value of F/W. With 16 engines, a F/W near the optimum (minimum launch weight) was obtained, but with 15 engines, minimum inert weight was obtained at a launch weight increase of only 30,000 lb. The 15-engine configuration was selected for the final FR-3 vehicle.

2.1.2 NOZZLE EXPANSION AREA RATIO

2.1.2.1 2 X 2 X 2 FR-1 Vehicle. Area ratio (ϵ) optimization studies were made for the 2 X 2 X 2 configuration for both two-position nozzles and fixed nozzles of the Pratt and Whitney "maximum performance contour" as defined in P&W FR-492D and discussed below. Results of these studies are summarized in Table 2-1, showing normalized launch weight at the optimum area ratios for the configurations studied. There is a substantial advantage (12.4 percent in launch weight) for the two-position nozzle compared with the fixed nozzle. The table also shows that use of identical two-position nozzles in all elements yields launch weights 3.7 percent lower than a configuration using optimized fixed nozzles in the booster and two-position nozzles in the orbiter.

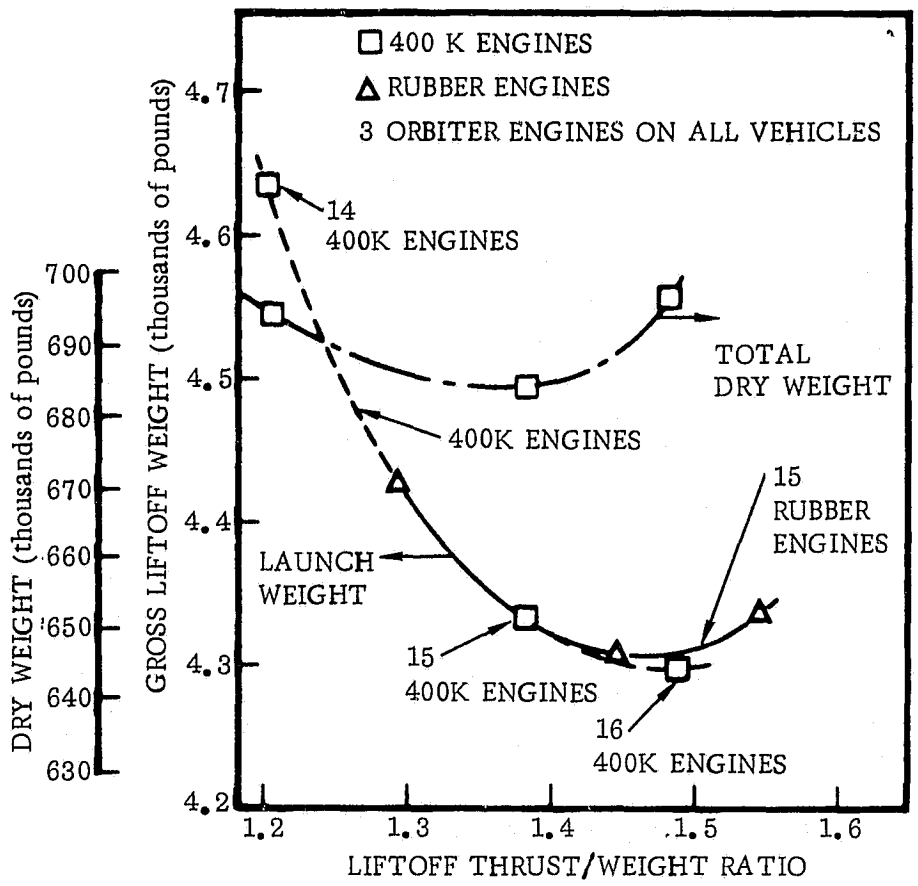


Figure 2-7. Effect of Number of Engines and F/W on FR-3 Performance

Table 2-1. Launch Weight Comparisons at Optimum Area Ratios

Two-Position Nozzles	
Identical Engines, All Elements	100 %
Two-Position Orbiter, Fixed Booster	103.7 %
Fixed Nozzles	
Identical Engines All Elements	112.4%
Different Orbiter/Booster Area Ratios	109.2%

Figure 2-8 summarizes results of studies in which two-position nozzles were considered. For the "all common" configuration (identical engines with two-position nozzles on all stages), launch weight decreased with ϵ to 200, the maximum value considered, but element inert weight and engine size were minimum at $\epsilon = 160$. For the "uncommon" configuration (fixed nozzles on boosters, two-position nozzles on orbiter), launch weight, thrust, and element inert weights all decrease with orbiter ϵ to 200. Launch weight decreases with booster ϵ to 60, but element inert weight is nearly flat and required thrust minimizes at $\epsilon = 40$.

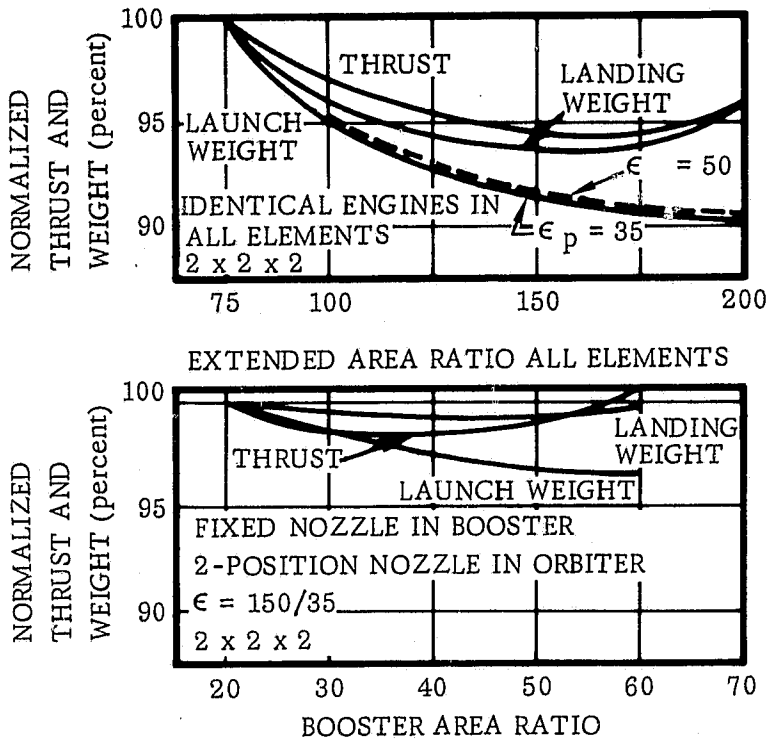


Figure 2-8. Area Ratio Two-Position Nozzles

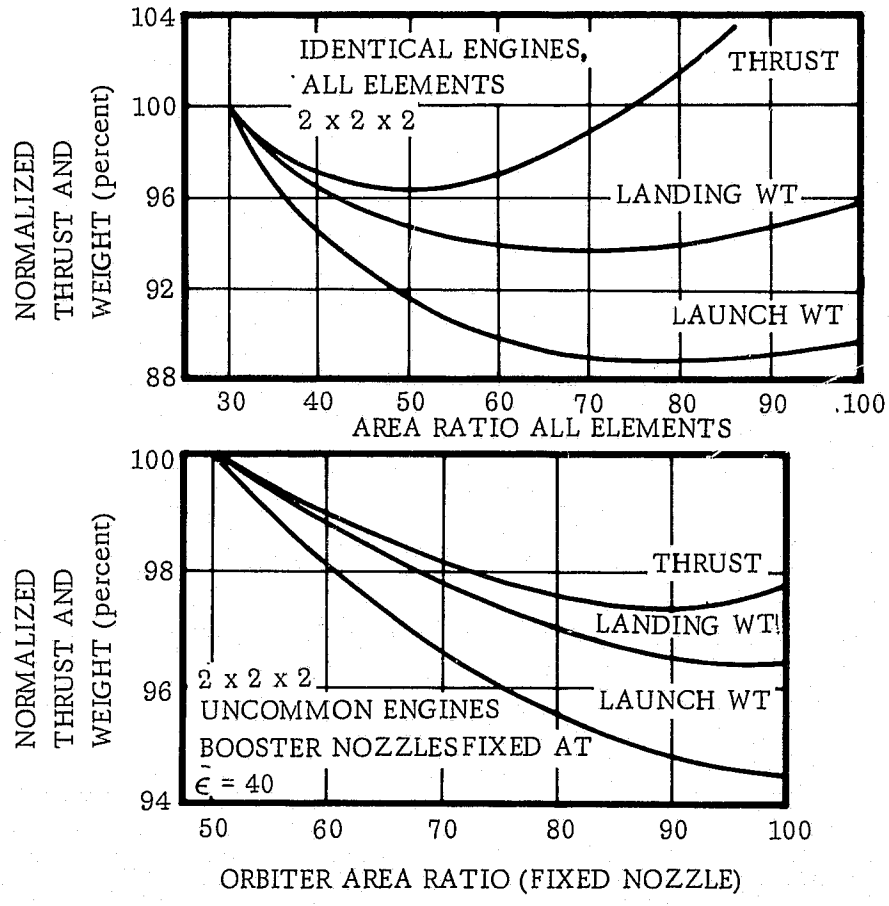


Figure 2-9. Area Ratio Fixed Nozzles

Figure 2-9 summarizes results of studies in which fixed nozzles were considered. For the all-common configuration, the launch weight, inert weight, and required thrust minimize at area ratios of 80, 70, and 45, respectively. The optimum value of ϵ would probably be about 55, where the inert weight is nearly minimum, at a very small increase in thrust compared to the minimum value. For the uncommon nozzles, launch weight, inert weight, and required thrust minimize at area ratios of 100, 95, and 88, respectively; the optimum value is about 90.

2.1.2.2 FR-3 Vehicle. A tradeoff study has been made to determine the optimum area ratios for the FR-3 two-element sequential burn vehicles for the 50,000-lb, 60 ft x 15 ft payload. The vehicle on which these studies were made used 16 booster engines and 3 orbiter engines. Engines of approximately 400,000-lb sea level thrust were used, but actual value was allowed to float to prevent variations in thrust/weight ratio, which could have masked area ratio effects. Results of this study are summarized in Figures 2-10 through 2-13. Three different engine configuration options were investigated. Results for the all-common configuration with identical two-position nozzle engines in orbiter and boosters are given in Figure 2-10. Results for the uncommon configuration, with two-position nozzles in the orbiter and fixed nozzle in the booster are given in Figure 2-11. These results are crossplotted in Figure 2-12 to show the effect of orbiter ϵ on launch weight. Results for the uncommon configuration with different two-position nozzles in the orbiter and in the boosters are given in Figure 2-13.

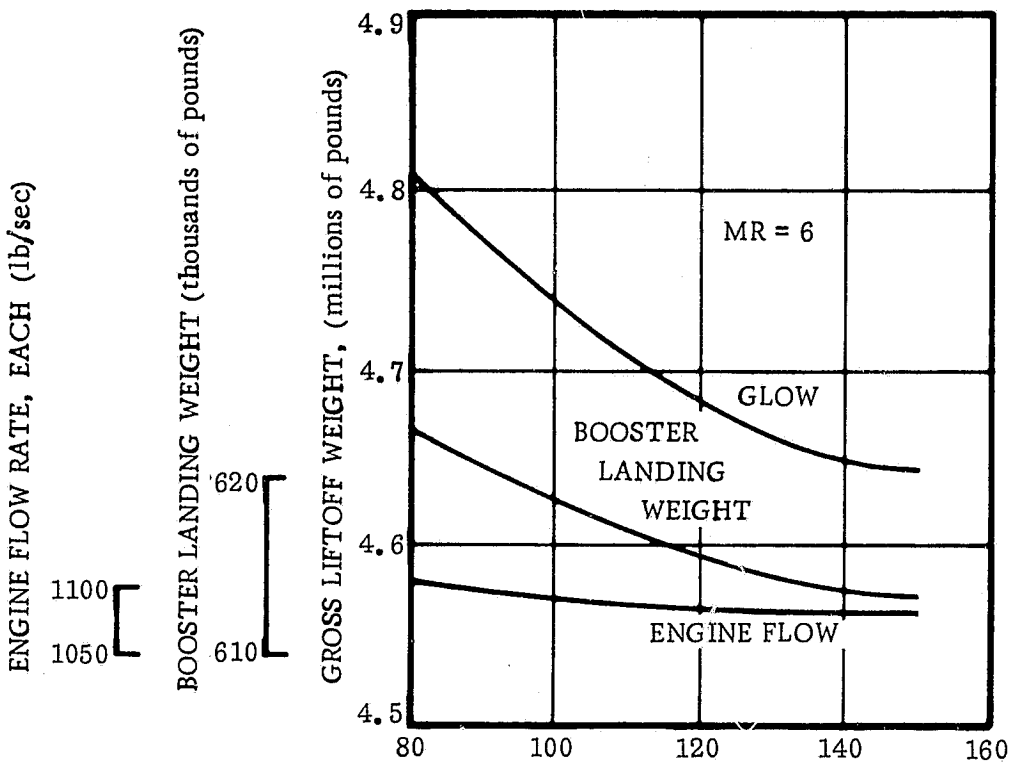


Figure 2-10. FR-3 Area Ratio Sensitivity with Two-Position Nozzles in Orbiter/Booster

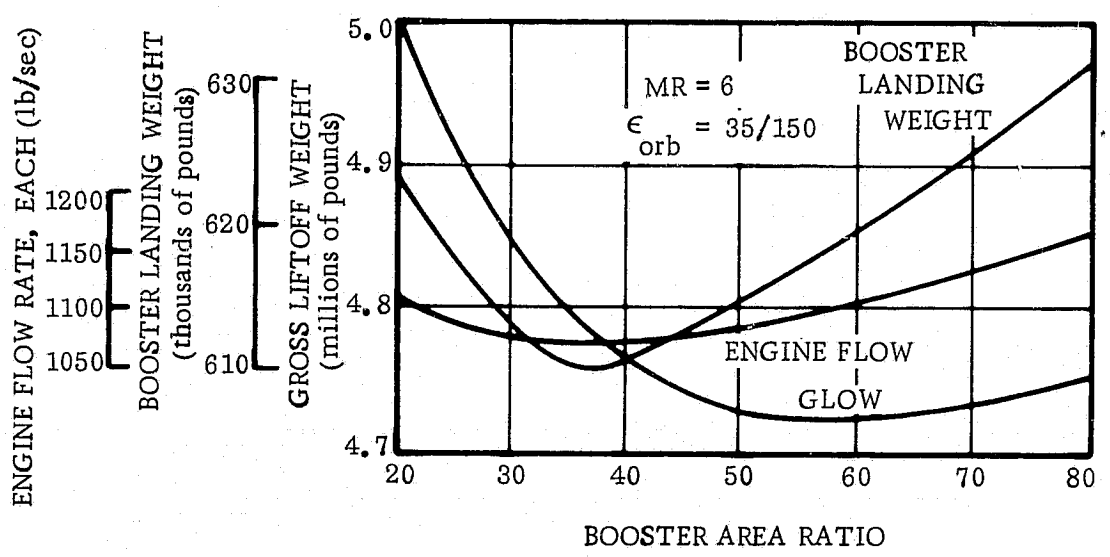


Figure 2-11. FR-3 Area Ratio Sensitivity with Fixed Booster Nozzles, Two-Position Orbiter Nozzles

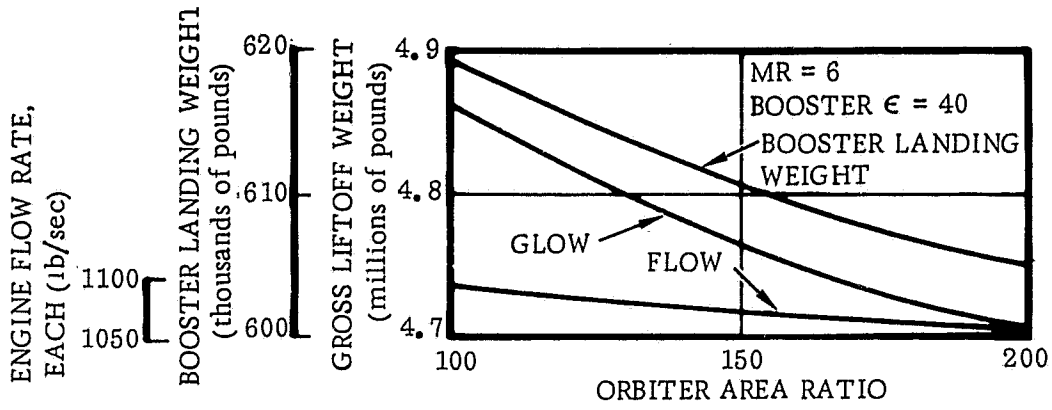


Figure 2-12. FR-3 Area Ratio Sensitivity with Fixed Booster Nozzles, Two-Position Orbiter Nozzles (Cross-Plot)

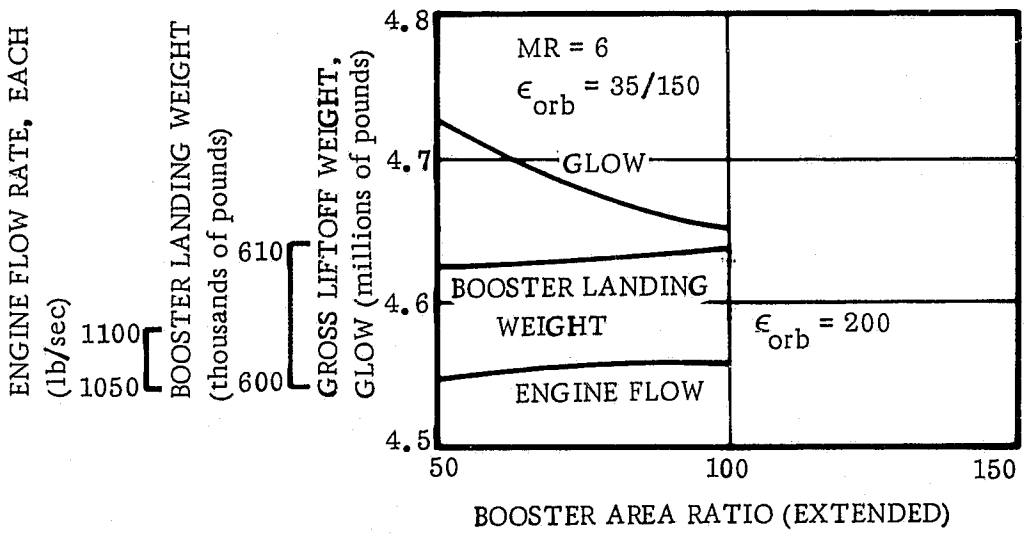


Figure 2-13. FR-3 Area Ratio Sensitivity with Two-Position Nozzles in Orbiter and Booster

Using common nozzles in both stages (Figure 2-10), a continual reduction in overall launch weight is obtained out to the maximum area ratio considered (150). Engine size (indicated by engine flow rate) and landing weights of orbiter and booster also follow a continual reduction. However, the maximum area ratio for which space is available in the booster is 100, so the orbiter would be limited to this value as well.

Using fixed booster nozzles (Figure 2-11), a definite minimum weight is obtained at a booster area ratio of 60. Orbiter landing weight also optimizes at $\epsilon = 60$, but engine size and booster landing weight are minimum at $\epsilon = 40$.

Using different two-position nozzles in orbiter and booster, launch weight and orbiter landing weight show a continual reduction out to a booster ϵ of 100, the highest value considered since large nozzles would not fit into the vehicle. Booster landing weight and engine size are nearly constant. The maximum value that will fit in the vehicle base should result in the lowest cost system.

Results of all the FR-3 area ratio tradeoffs are summarized in Table 2-2 which shows launch weight and engine size at the selected optimum area ratios for each nozzle combination considered. This study has shown a significant performance improvement for use of orbiter area ratios up to 200. A lower booster area is required, with little difference between an optimum fixed nozzle of $\epsilon = 40$ and the largest two-position nozzle for which room is available ($\epsilon \cong 100$). In either case, two different nozzles are required.

Pratt and Whitney has estimated that development of two different nozzles for the 400,000-lb thrust engine would increase development program cost 15 percent. However, if the nozzles were designed so that a common primary nozzle (fixed portion of two-position nozzle) could be used with two different movable extensions, cost would increase by only five percent (compared to a single nozzle). Such a nozzle would have to be a compromise and would have lower performance under some conditions.

A configuration was investigated in which the nozzle is optimized for the booster, using the 35/100 combination. For the orbiter a compromise extension having the same primary and giving an area ratio of 200 is used with a loss in I_{sp} of 0.5 percent or 2.3 seconds. Performance and cost were established for two otherwise identical vehicles, one using this compromise 35/100-200 nozzle combination and one using optimum 35/100 and 35/200 nozzles. The results are summarized in Table 2-3. Use of the compromise nozzle increased vehicle launch weight 50,000 lb. However, the saving in engine development cost more than offset the increased vehicle cost which resulted, for a net saving of $23 million in program cost.

2.1.2.3 Nozzle Gimbal Clearance and Fairing Considerations. This section summarizes results of a study of the relation of engine area ratio, nozzle clearance, and fairing requirements for different assumptions concerning the method of accommodating a thrust vector control (TVC) failure:

a. "Idiot Proofing." Engines are located so that it is impossible for nozzles to touch, regardless of engine gimbal position. A TVC engine can fail in any position and not interfere with gimbaling of remaining engines.

b. Null Seeking. An engine with inoperative TVC will automatically move to a desired position. Engines are located to take advantage of this location.

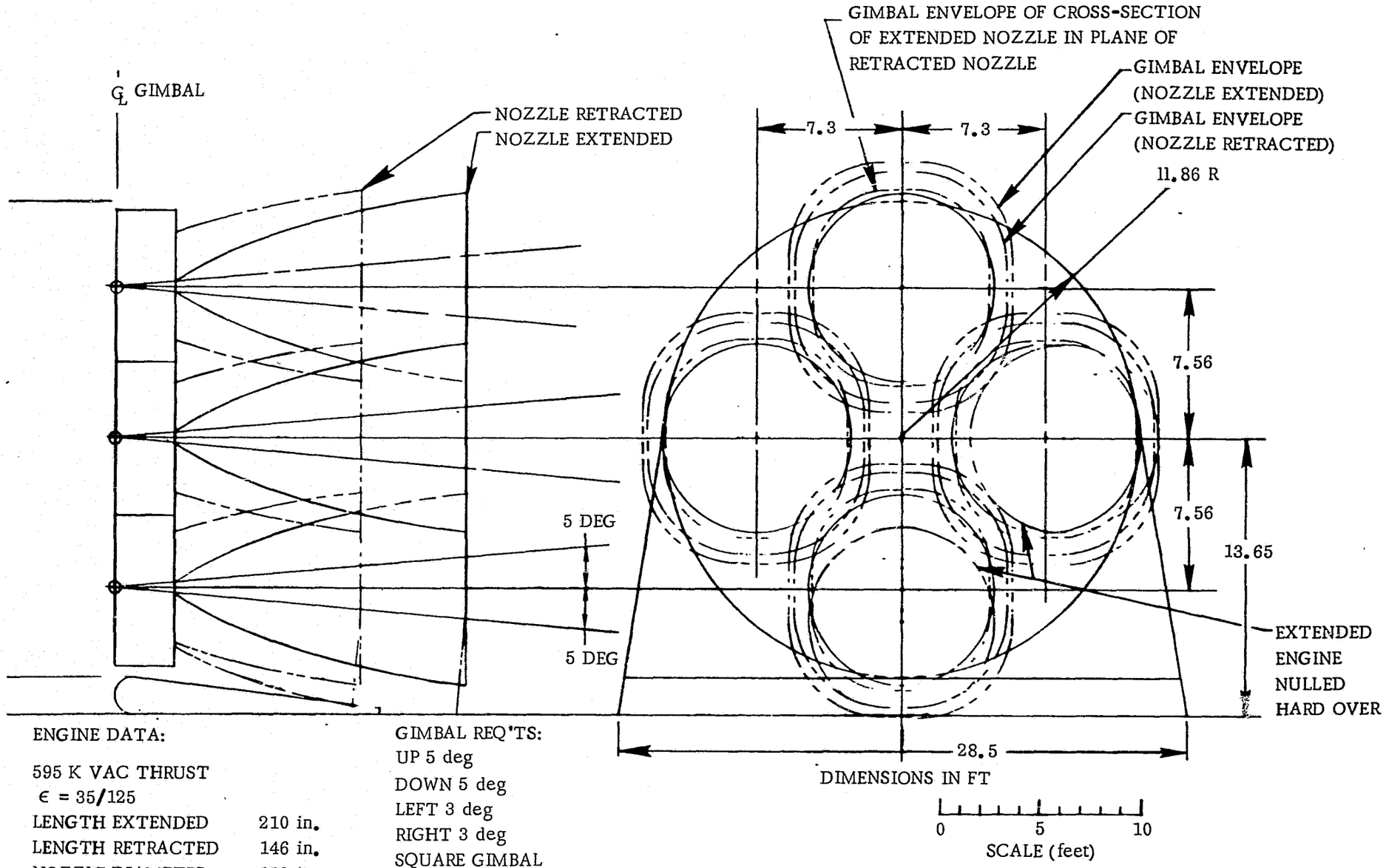
Both of these assumptions were evaluated for the two-position nozzles, extended and retracted. The vehicle engine arrangement assumed for this study is shown in Figure 2-14. Gimbal angle requirements considered were ±5 degrees vertically and ±3 degrees horizontally. With the four engine gimbal blocks fixed in the locations shown, minimum clearance between the engines was determined for each failure mode assumed. Results are given in Figure 2-15 in the form of nozzle gimbal clearance versus area ratio for six failure conditions. Permissible area ratios for the configuration studied vary from approximately 70 for idiot-proof clearance with nozzles extended to 120 for null-to-center with nozzle retracted on the failed engine.

Table 2-2. Comparison of Nozzle Type at Optimum Area Ratios

Two-Position Nozzles	Area Ratio Booster/Orbiter	Launch Weight (lb)	Engine Flow Rate (lb/sec)	Relative Engine Development Cost (percent)
Identical Engines, All Elements	100/100	4,740,000	1092	100
Optimum Nozzles, Each Element	100/200	4,596,500	1058	115
Fixed Nozzles				
Optimum Nozzles, Each Element	40/200	4,705,000	1060	115

Table 2-3. Two-Position Nozzle Configuration Comparisons

	Area Ratio Booster/Orbiter	I_{sp} VAC Booster/Orbiter	Launch Weight (lb)	Δ Engine R&D Cost	Δ Program Cost ($)
Optimum Nozzles Each Stage Common Primary, Different Secondary Nozzles	See Confidential Addendum				



ENGINE DATA:

595 K VAC THRUST
ϵ = 35/125

LENGTH EXTENDED	210 in.
LENGTH RETRACTED	146 in.
NOZZLE DIAMETER	113 in.
POWER PACK DIAMETER	92 in.

GIMBAL REQ'TS:
UP 5 deg
DOWN 5 deg
LEFT 3 deg
RIGHT 3 deg
SQUARE GIMBAL PATTERN

Figure 2-14. Engine Gimbal Clearance Study Baseline Configuration

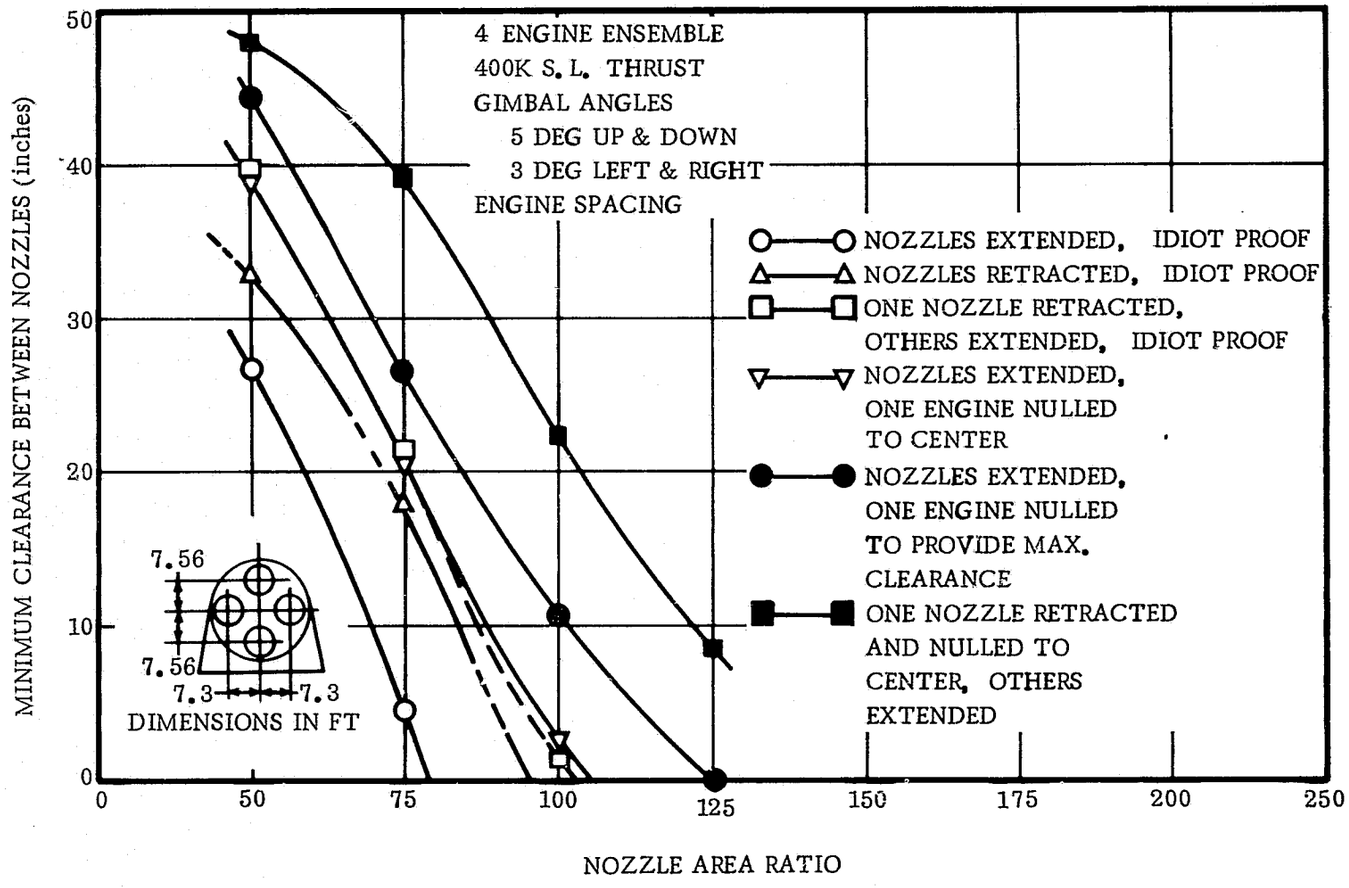


Figure 2-15. Engine Clearance vs Expansion Ratio

Projected fairing areas required for expansion ratios of 35/125, 35/100, and 35/75 for the reference configuration are shown in Figure 2-16. The expansion ratio would have to be reduced to 35/50 to completely eliminate fairing requirements.

2.1.3 MIXTURE RATIO. Mixture ratio (MR) trade studies made for the 2 x 2 x 2 FR-1 are summarized in Figure 2-17. This study was made assuming the characteristics of an engine designed to operate over an MR range of 5.0 to 7.0. Use of lower MR values improves specific impulse (9.2 seconds higher at 5.0 than at 7.0 for the configuration studied) at the expense of lower propellant bulk density, which increases vehicle size for a given propellant load. A definite optimum was found at MR near 7.0, as indicated in Figure 2-17. This study was made assuming a constant area ratio. Refinement of this study to include effects of reduced area ratio at higher launch weights would degrade performance (increase launch weight) at MR on either side of 7.0, increasing the sharpness of the optimum.

A study was made to determine the mixture ratio excursion around the nominal which is required for propellant utilization (P.U.) control for the 2 x 2 x 2 FR-1. Results of this study are shown in Figure 2-18 giving allowable engine operating MR error as a function of P.U. control capability provided. It was assumed that all propellant tanks are calibrated to minimize unknown volume errors and that knowledge of systematic system errors is based on experience from a pre-operational and operational flight test history of six flights. A control capability of 5 percent should be adequate for requirements of the FR-1, which for a nominal MR of 7.0 would be 7.0 $\pm$0.35.

A similar study was made for the FR-3 two-stage sequential vehicle. The same results were obtained as for the FR-1 (optimum mixture ratio = 7 for both orbiter and booster). In addition, unequal mixture ratios between the orbiter and booster were investigated, with the finding that the vehicle is less sensitive to booster than orbiter M.R., as indicated by the top and bottom curves of Figure 2-19, where orbiter M.R. is held constant.

2.1.4 NOZZLE CONTOUR. All of the basic trade studies described in the previous sections have used a nozzle with a contour designated by Pratt and Whitney as the "maximum performance" contour which gives the highest specific impulse practicable. Pratt and Whitney also provided data on a "minimum surface area" nozzle (which gives minimum engine weight), a "minimum length" nozzle, and a "base" nozzle with length between that for the minimum surface area and the maximum performance contours. Use of these contours was investigated for the 2 x 2 x 2 engine configuration. Results are summarized in Table 2-4, showing normalized launch, inert, and engine weights and engine length for each contour. There is a substantial improvement in engine length and weight at a small (0.32 percent) increase in vehicle weight by use of the base nozzle contour. Engine weight is decreased 2 percent, which moves the vehicle center of gravity in the favorable direction. Smaller improvements in engine length and weight are obtained with the minimum length and minimum surface area nozzles,

Figure 2-16. Fairing Areas Required for ϵ = 35/125, 35/100 and 35/75

Table 2-4. Comparison of Performance, Weight, and Engine Length for Pratt and Whitney Nozzles (2-2-2, ϵ = 35/100)

Nozzle Type	Launch Weight	Inert Weight	Engine Weight	Length	I_{sp} SL/VAC
Maximum Performance Base Minimum Weight Minimum Length	See Confidential Addendum				

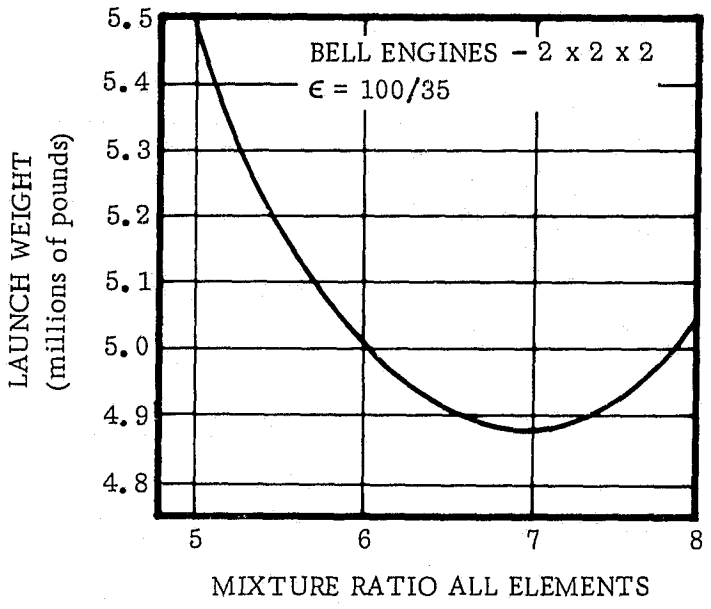


Figure 2-17. Mixture Ratio

at a larger penalty in vehicle weight. Use of the base nozzle contour would appear to be the best compromise for FR-1 because of requirements for protecting the nozzle from entry heating.

2.1.5 TAKEOFF RATING OPTION. The initial engine requirement called for operation over a mixture ratio band of 5.0 to 7.0 at thrusts between 20 percent and 100 percent, as shown in Figure 2-20. The requirement to operate at the MR of 5.0 at 100 percent thrust sizes the turbomachinery, and operation at higher thrust at higher MR is possible. Thermal limits for the engine are established by the requirement to operate at the MR of 7.0 at 100 percent thrust, and operation at higher thrust at lower MR is possible. Maximum thrust capability of the engine is defined by the intersection of the turbomachinery and thermal limit lines shown in Figure 2-20. This point was defined as 114 percent of vacuum thrust (115 percent at sea level) at MR = 6.4. To operate at this point would require oxidizer net positive suction pressure (NPSP) 2 psi above nominal, which is greatly exceeded by elevation head pressure in the FR-1.

The engines for FR-1 are sized by the liftoff thrust requirement. Typically, liftoff thrust is maintained until the F/W limit of 3.0 is reached, then the engines are throttled to maintain 3.0. Beyond this point, high thrusts are not required. This suggests that the engine could be operated during the early phases of flight at the maximum thrust capability (114 percent thrust and MR = 6.4), then throttled to lower thrust and

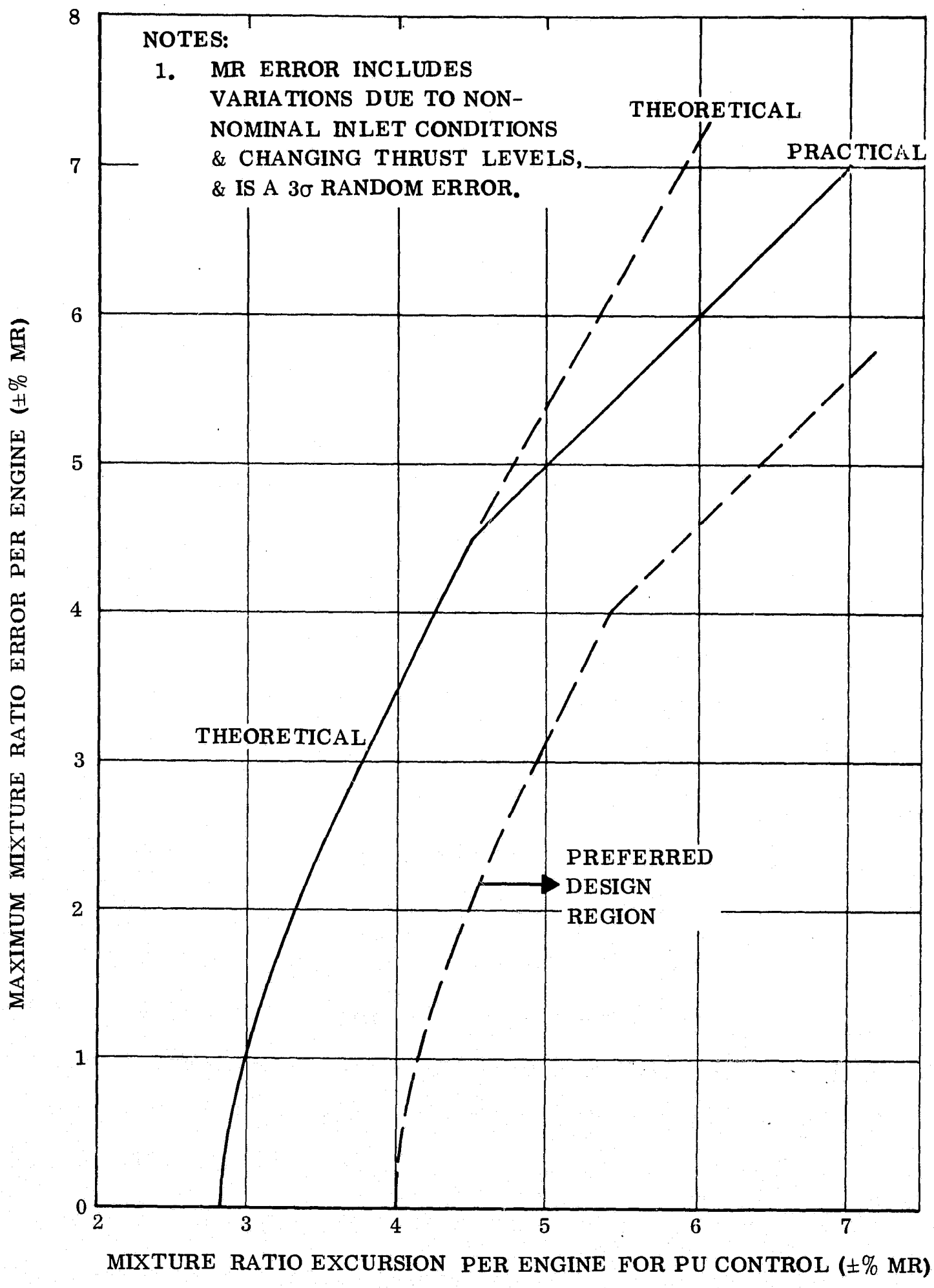


Figure 2-18. Allowable Engine MR Error vs. MR Excursion for Propellant Utilization Control

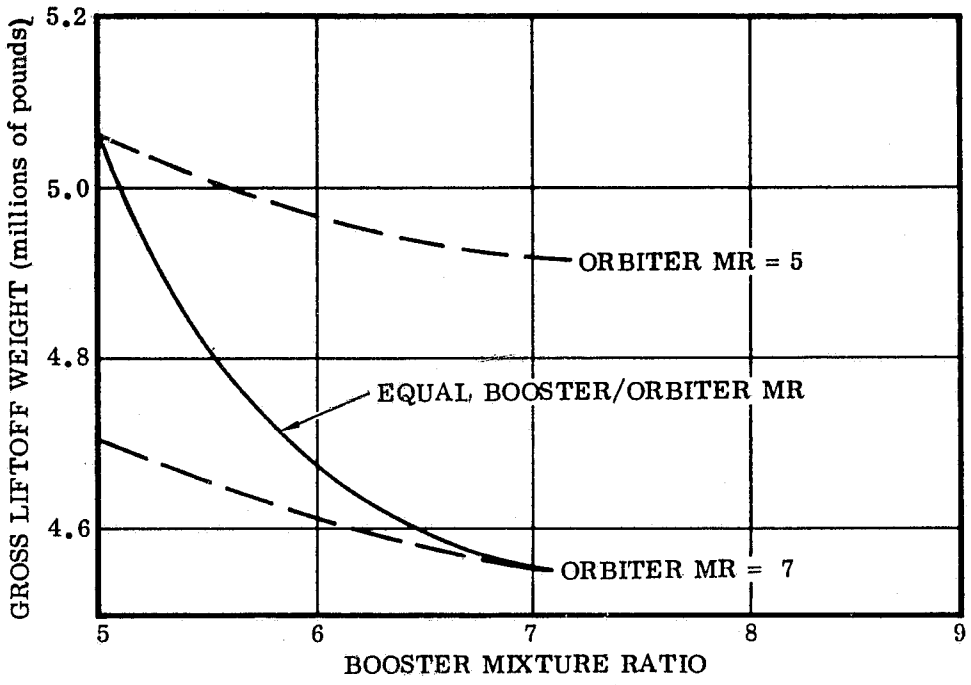


Figure 2-19. Mixture Ratio Tradeoff

Figure 2-20. Engine Operating Characteristics

the desired MR of 7.0 for the latter part of boost phase and all of the orbiter solo phase. This would yield most of the benefits of operation at the optimum MR yet obtain the performance gains due to operation at maximum thrust at launch. These gains are due to the 14 percent increase in thrust (which effectively reduces engine specific weight 14 percent), increased sea level I_{sp} due to operation at a higher pressure ratio, and increased vacuum I_{sp} because a higher area ratio can be fitted into a limited base area at the higher chamber pressure. The area ratio effect is by far the most significant of the three.

A study was made to compare operation at the maximum thrust capability (a liftoff rating) at launch with operation at the nominal 100 percent level at MR = 7.0. Because of computer program limitations, the MR of 6.4 required for the liftoff rating was maintained through booster staging. Table 2-5 summarizes results of this study. Vehicle launch weight is reduced 4.6 percent and engine design vacuum thrust at the nominal rating is reduced 16.8 percent.

Table 2-5. Effect of Liftoff Rating Option

Liftoff Conditions	
Thrust	115% of nominal
Mixture Ratio	6.4
Effects	
Area Ratio Increased	20%
Sea Level I_{sp} Increased	1.3% (5 seconds)
Launch Weight Reduced	4.6%
Design Vacuum Thrust Reduced	16.8%

2.1.6 CROSSFEED VS. NO CROSSFEED

2.1.6.1 Vehicle Considerations. With many of the parallel-staged vehicle configurations being considered for space shuttle, it is possible and from some standpoints desirable to burn all engines of both booster and orbiter at launch. Operation of all engines at full thrust throughout boost phase is possible if crossfeed plumbing is provided to permit operation of the orbiter engines on propellants supplied from the booster. If all engines operate at full thrust during boost phase, the following advantages are obtained:

a. Since the orbiter engines provide part of the required thrust, thrust required by the booster is reduced, reducing either the size or number of booster engines. Reduction in engine weight improves vehicle performance. Depending on the specific vehicle configuration, use of a higher expansion ratio or a reduction in

size of aerodynamic fairings required for nozzle protection may be possible, further improving performance.

b. Loads on the connections between vehicles are reduced, since the orbiter, instead of being carried inert up to the maximum boost phase acceleration (generally 3 g), is being accelerated by its own engines (generally at about 1.5 g). This could reduce interconnection loads by up to 50 percent, depending on design criteria for orbiter engine out. This reduction in weight could improve vehicle performance.

c. All main engines operate from launch, eliminating the requirement for in-flight start of orbiter engines. This is somewhat more important for the space shuttle than for past rocket vehicles because of the intact abort requirement (no separate escape system for crew or passengers). Complete failure of the orbiter engines to operate results in loss of vehicle and crew.

d. Balance of booster vehicles is improved because of reduction in engine weight.

Addition of subsystems required for propellant crossfeed capability in the space shuttle produces a number of disadvantages:

a. Hardware to interconnect the main propellant subsystems of the booster and orbiter must be added. At the minimum, a main propellant crossfeed incorporating a retracting, nonspilling disconnect 10 to 20 inches in diameter is required for each propellant. For additional safety, valves to back up the disconnect, recirculation systems, and purge systems may be required.

b. For some vehicle designs, the crossfeed plumbing must penetrate the heat shield, requiring doors and related opening/closing mechanisms. Failure of a heat shield door could result in loss of a vehicle.

c. At staging, orbiter propellant flow must be phased from booster tanks to orbiter tanks, crossfeed lines disconnected and retracted, and heat shield doors closed. Timing of these functions is critical and may impose some performance penalties.

d. Component development programs are required for crossfeed hardware. An all-up system static firing test stand coupling the orbiter and the booster may be required.

e. Pressure transients at booster engine shutdown and at staging may affect operation of the orbiter engine.

Because of the significant disadvantages associated with crossfeed, a study was made to determine the performance of comparable vehicles with and without crossfeed.

2.1.6.2 Performance Comparison. The crossfeed comparison study was made on three-element vehicles, those with crossfeed designated FR-1 and those without designated FR-4. The procedure used for the study was to synthesize vehicles with and without crossfeed for identical missions defined in Table 2-6, using the Convair space shuttle synthesis program incorporating a fixed weight penalty for the crossfeed

Table 2-6. Mission Requirements for Crossfeed Study

Payload	50,000 lb
Launch Site	ETR
Orbit Inclination	55 deg
On-Orbit ΔV	1800 fps
ACS ΔV	200 fps
Launch Thrust/Weight	1.47
Maximum F/W	3
Staging Dynamic Pressure	50 psf
Injection Altitude	45 n. mi.
Final Circular Orbit Altitude	100 n. mi.

plumbing, residuals, and heat shield doors. Estimated weight penalties generated by NASA MSFC and Convair were considered. Initially, the weight penalties used were based on 4-4-4 configurations using 500,000-lb-thrust bell engines for both the FR-1 and FR-4. Performance generated is shown in Table 2-7 and Figure 2-21. Two major conclusions were drawn:

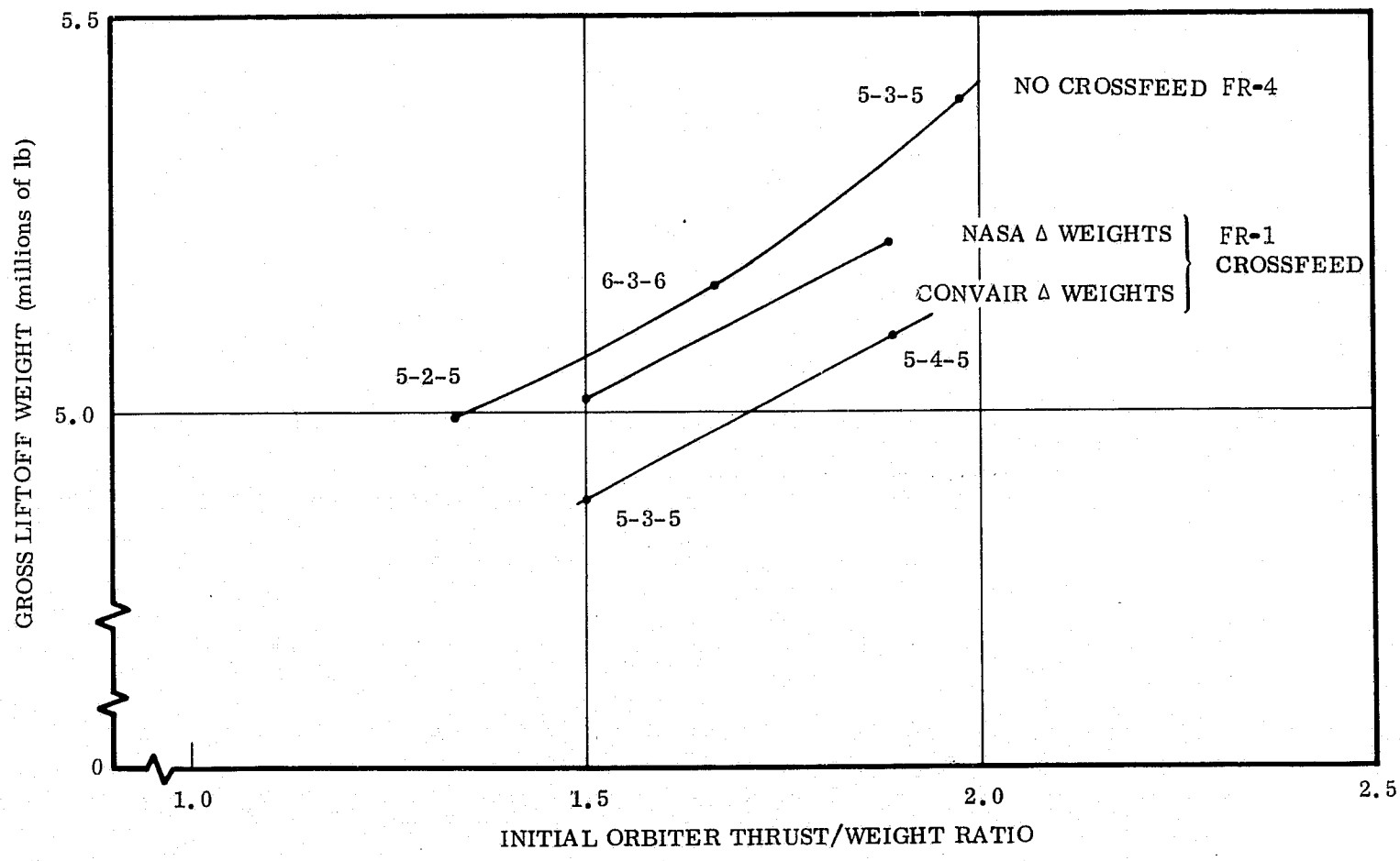


Figure 2-21. Launch Weight Comparisons for FR-1 and FR-4 Vehicles

Table 2-7. Crossfeed Comparison, 50K Pound Payload Vehicle, Various Engine Arrangements

	No Crossfeed	No Crossfeed	No Crossfeed	Crossfeed NASA ΔWT(1)	Crossfeed GDC ΔWT(1)	Crossfeed NASA ΔWT(1)	Crossfeed GDC ΔWT(1)
Engine Arrangement	5-3-5	6-3-6	5-2-5	5-3-5	5-3-5	5-4-5	5-4-5
Liftoff Weight — lb Booster Sea Level Thrust/Engine — lb Orbiter Vacuum Thrust/Engine — lb Orbiter I_{SP}, S.L./VAC Booster I_{SP}, S.L./VAC Booster Engine Weight — lb Orbiter Engine Weight — lb Orbiter F/W @ Staging Liftoff Weight Change — lb	See Confidential Addendum						

(1) Added weights for crossfeed are as follows:

	NASA	Convair
Orbiter	6412	2845
Booster (2)	6652	7028

(2) This orbiter F/W ratio is below the minimum required for "once around" abort in event of one orbiter engine failure (1.54 minimum with a total of 2 engines).

Note: Liftoff F/W for all vehicles is 1.47.

a. For a given configuration (for example, a 5-3-5 engine arrangement) use of crossfeed reduces launch weight between 8 and 11 percent.

b. There are different optimum engine arrangements for FR-1 and FR-4, and use of these optimum configurations minimizes performance differences. Vehicle performance was found to be a strong function of initial orbiter F/W ratio. For equal volume vehicles with common orbiter and booster engines, orbiter F/W is uniquely determined by the engine arrangement (i.e., relative number of engines in orbiter and boosters). For the FR-4, orbiter thrust/weight is substantially higher with a given engine arrangement than for FR-1. For vehicles of equal weight and a 5-3-5 engine arrangement, FR-4 orbiter thrust is 30 percent of launch thrust (3/5+5) while for FR-1 if is only 23 percent (3/5+5+3), so FR-4 initial orbiter F/W is about 30 percent higher. To equalize orbiter F/W ratios, the number of orbiter engines divided by the total number of engines operating at liftoff should be approximately the same.

Because of the importance of the engine arrangement in a proper evaluation of crossfeed, weight penalties for crossfeed were reassessed with more nearly optimum engine arrangements. Propellant feed designs were developed for the 6-3-6 FR-4 and the 5-3-5 FR-1 described in Table 2-7. These designs are shown in Figures 2-22 and 2-23. Two crossfeed arrangements are shown, one with "full" manifolding which allows flow of propellant completely across the vehicle from one booster to the other, and one with "partial" manifolding which provides the minimum connections required for feeding the orbiter engines. The latter arrangement has the same operational disadvantage as the no-crossfeed case in that booster-engine-out results in a build up of cg asymmetry because of unequal booster propellant consumption.

Propellant subsystem weight differences between the no crossfeed 6-3-6 arrangement and the two crossfeed arrangements were then determined. These differences are summarized in Table 2-8. Orbiter feed subsystem weights are substantially increased because of added disconnects, shutoff valves, TPS doors, and residuals in the added ducting. Booster feed subsystem weights are almost unchanged with full manifolding, and reduced 1000 lb with partial manifolding. The reason for this reduction is not obvious from inspection of the feed subsystem layout. However, when it is considered that each FR-4 booster has six engines of 631,000 lb thrust, while the FR-1 has only five engines of 543,000 lb, each with required plumbing and residuals, the reasons become apparent.

Figure 2-24 was then developed, using the crossfeed weight penalties given in Table 2-8. Gross liftoff weight (GLOW) for FR-4 no-crossfeed vehicles and FR-1 crossfeed vehicles with full and partial manifolds are shown as a function of orbiter F/W ratio. Engine arrangements which result in the parametric F/W values are indicated. Evaluation of the abort capabilities of these configurations with one orbiter engine out shows that the 5-3-5 FR-1 is the lightest satisfactory configuration. The 8-3-8 FR-4 configuration is satisfactory, since orbiter engine loss effects only the orbiter solo phase, and not the boost phase of flight.

Table 2-8. Added Weights for Crossfeed for 5-3-5 FR-1 Compared with 6-3-6 FR-4 (50,000 lb Payload)

	LO_2 System	LH_2 System	Total
Full Manifolding			
Orbiter	6980	1148	8128
Booster (each	510	-437	73
Partial Manifolding			
Orbiter	5820	1148	6968
Booster (each)	-560	-437	-997

NOTE: Plus weight values mean crossfeed system weights heavier, minus values mean crossfeed system weights lighter.

The following conclusions may be drawn from the data presented in Figure 2-24.

a. For a given combination of orbiter and booster engines, use of crossfeed reduces vehicle launch weight. Data was not developed in this study for a completely "common" element vehicle (i.e., with the same number of engines in orbiter and booster), but the trend can be seen by comparing the FR-1 and FR-4 with the 5-3-5 engine arrangements. Launch weight was 6 to 9 percent lower for the FR-1 with crossfeed. Differences would be greater for common elements, since the curves of launch weight are diverging at the orbiter F/W ratios compatible with the common engine arrangement.

b. For vehicles optimized for either crossfeed or no crossfeed, differences in launch weight are minor, the exact differences depending on relative detail weight differences, pro and con, for crossfeed. This can be seen by comparing the 8-3-8 FR-4 configuration at 4.82 million pounds and the 5-3-5 FR-1 at 4.9 to 5.02 million pounds.

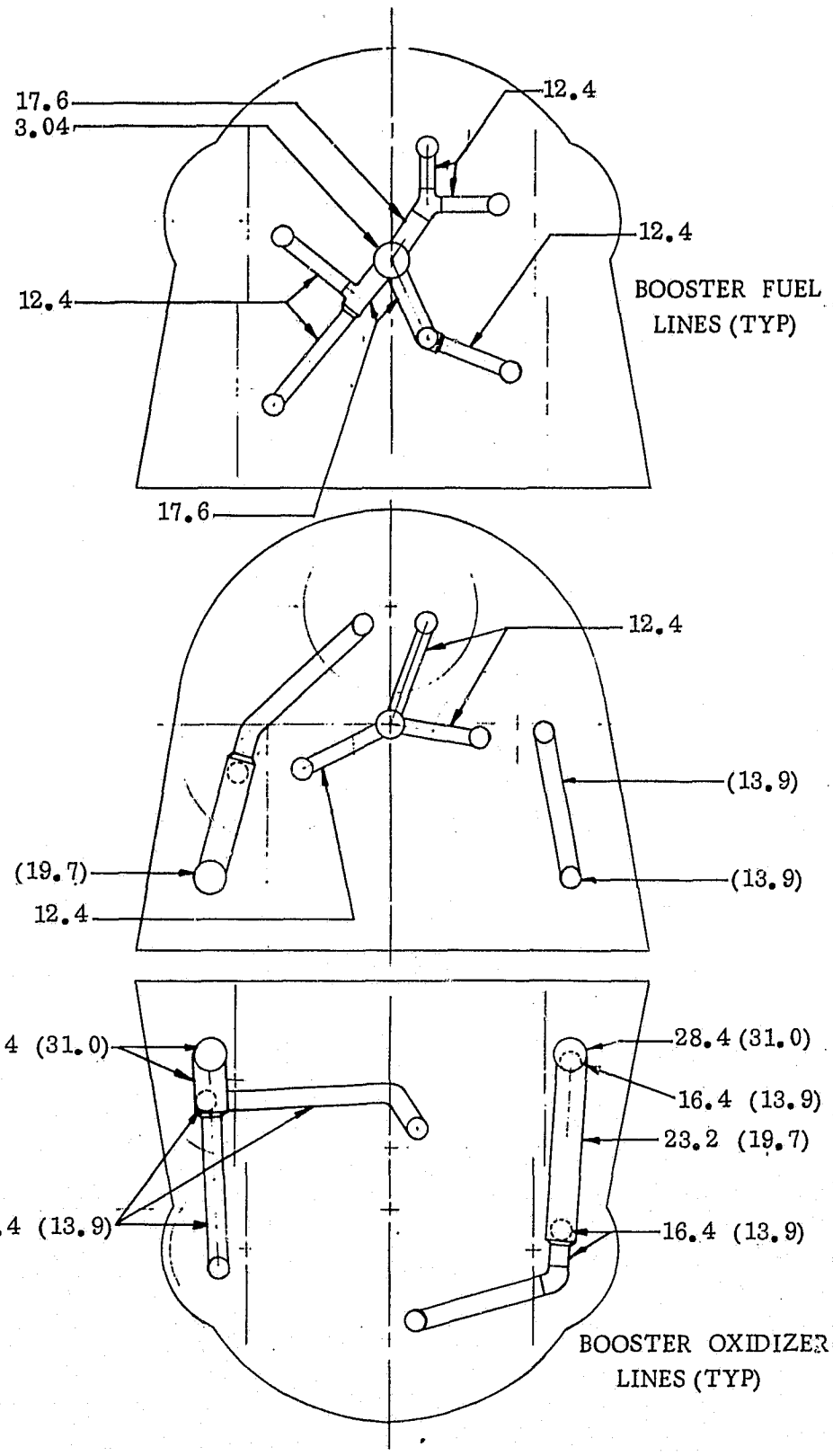
2.1.7 ORBIT MANEUVER PROPULSION

2.1.7.1 Introduction. In this section, several approaches for performing the required orbit maneuvers using main propulsion are compared. The vehicle used in performing the orbit maneuver studies is the FR-1 orbiter element having two one-million pound thrust main engines. During the space shuttle study effort, as the vehicle was modified and as better and more complete engine data became available, the comparisons were updated and additional approaches were considered.

2.1.7.2 Wet Pump Mode Comparisons. The initial comparison is reported in Reference 2-1. Values in that comparison are based on preliminary estimates of the idle mode specific impulse and propellant boiloff losses that have since been modified.

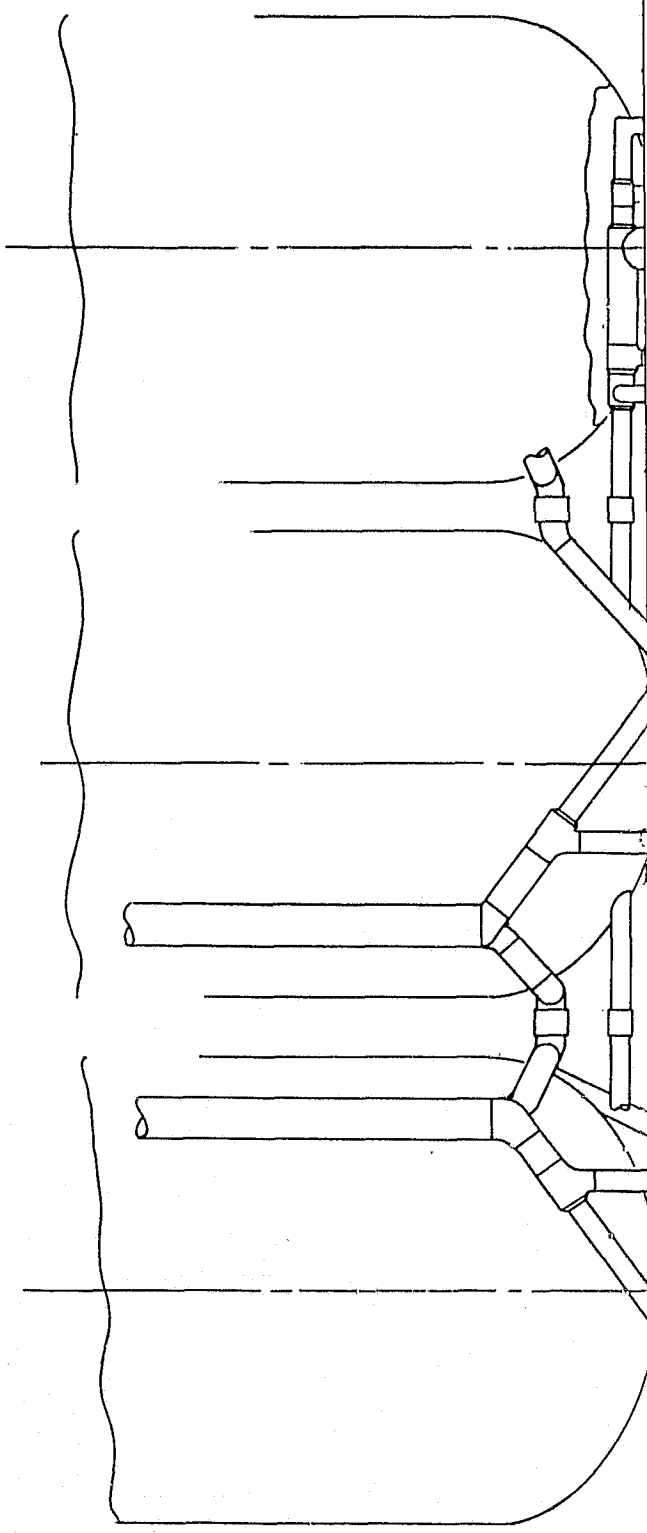
FR-4

6-3-6 CONFIGURATION
NO CROSSFEED



FR-1

5-3-5 CON
WITH CROS
FULL MAN



FOLDOUT FRAME

FR-1

5-3-5 CONFIGURATION WITH CROSSFEED AND FULL MANIFOLDING

NOTE:
1. NUMBERS REFER TO LINE DIAMETER SIZED FOR MAXIMUM FLOW ACCELERATION OF .004 LB/SEC^2 LB PER ENGINE.
2. NUMBERS IN ( ) REFER TO LINE DIAMETER SIZED FOR 20 FT/SEC IN LOWER MANIFOLD LINES.
3. ALL DIMENSIONS IN INCHES

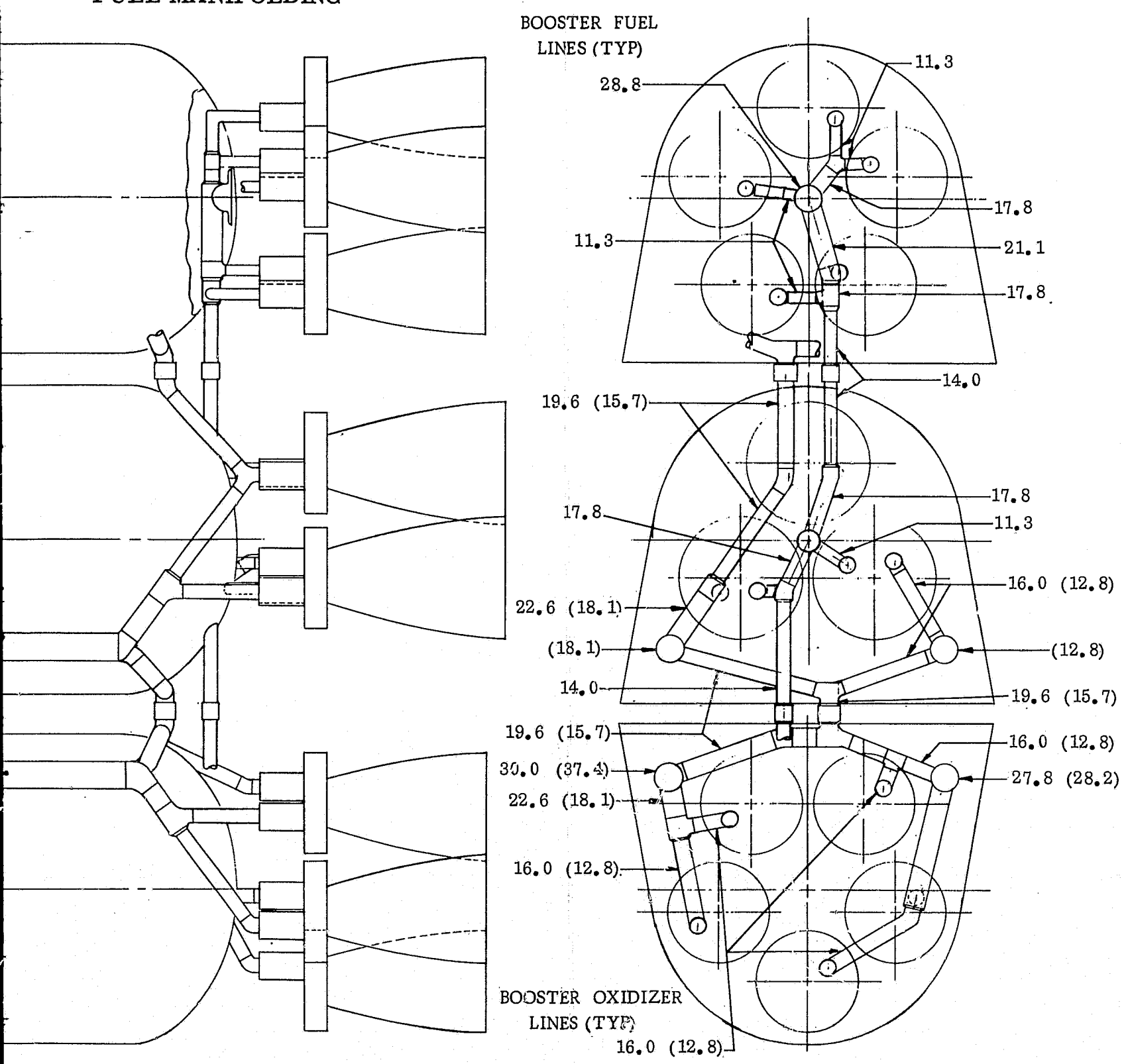
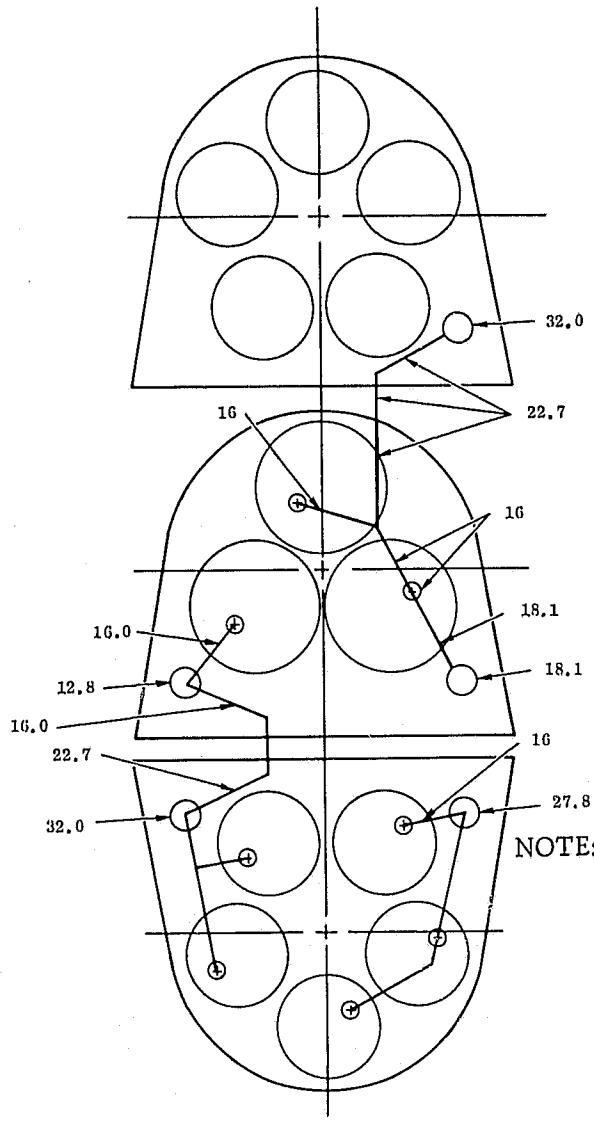


Figure 2-22. Propellant Feed Duct Routing for FR-1 and FR-4 Vehicles

FOLDOUT FRAME 2



NOTE: FUEL DUCTING IDENTICAL TO THAT SHOWN IN FIGURE 2-22. SEE NOTES ON THAT FIGURE.

Figure 2-23. FR-1 Propellant Feed Duct Routing — Partial Manifolding

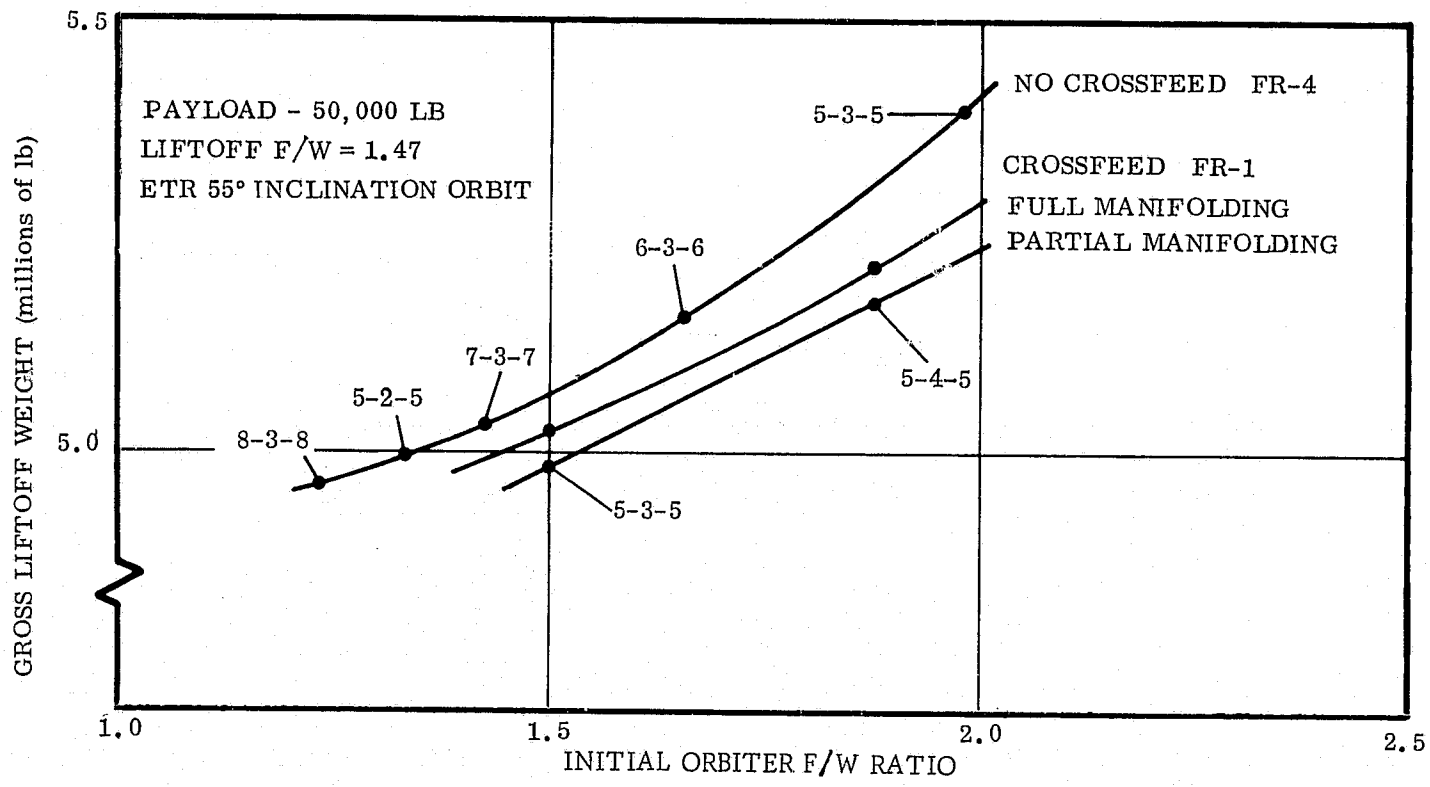


Figure 2-24. Launch Weight Comparison for FR-1 and FR-4 Vehicles

The comparison is repeated, therefore, using the modified values. Four approaches were compared. In the first approach, the standard XLR-129 characteristics were assumed with a minimum thrust of 20 percent of nominal, no idle mode, wet pumps, and tanks vented at 30 psia. Propellant settling was performed by an earth-storable reaction control subsystem (RCS). In the second approach, it was assumed that the engines had pressure-fed idle mode capability, that all orbit maneuvers were performed in idle mode, pumps remained wet, and tanks vented at 30 psia. In the third approach, idle mode was used for propellant settling, a 10 percent thrust pump-fed mode was used for the remainder of the maneuver, pumps remained wet and tanks vented at 30 psia. In the fourth approach, idle mode was used to perform the orbit maneuvers. This approach differs from the second approach in that the tanks were designed to withstand the pressure caused by heat soak-back from the engines rather than being vented at 30 psia.

The specific impulse for each mode of operation is shown in Table 2-9. The MR of the main engines for orbit maneuvers is 5.0 since this produces the maximum specific impulse. There are no significant advantages for the increased density at higher mixture ratios since adequate space is available for maneuver tanks without increasing vehicle volume.

Table 2-9. Maneuver Propulsion I_{sp}

Thrust/Engine	Pump Operating	I_{sp}
20 percent nominal	yes	See Confidential Addendum
10 percent nominal	yes	
Idle*	no	
3,500-lb (RCS)	—	

*Figure 2-25 shows idle mode thrust as a function of engine inlet pressure.

The mission profile assumed for this analysis is shown in Figure 2-26. This includes a 45-minute coast from the 45 n.mi. injection point to 100 n.mi., an engine firing to circularize at 100 n.mi., a phasing orbit of up to 17.5 hours, a Hohman transfer maneuver to a 260 n.mi. orbit, circularization, a coast for 149 hours, then a deorbit maneuver for entry.

Heat soak-back after firing and solar heat addition to the engine result in significant heat input to the orbit maneuver propellant. For the study, the engine was assumed to be oriented toward the sun during half of the coast period. Figure 2-27 shows the propellant losses if this heat is allowed to result in propellant boiloff. The losses are based on data furnished by Pratt and Whitney Aircraft for a 400,000 pound thrust engine that has been scaled by the ratio of the thrust levels. The alternate approach is to allow the heat input to increase tank pressure, as in the fourth approach.

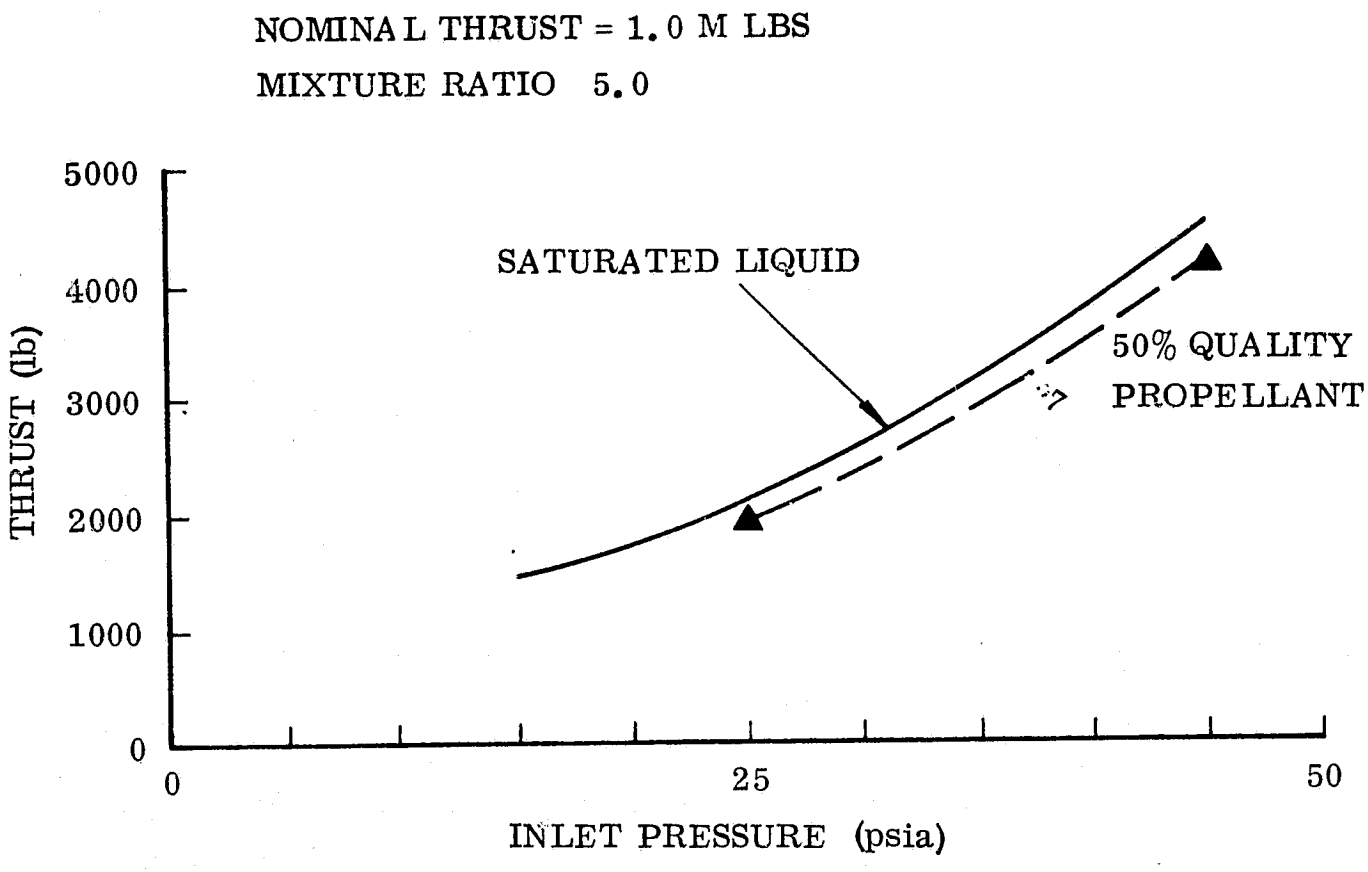


Figure 2-25. Idle Mode Thrust as a Function of Engine Inlet Pressure

For both of the approaches incorporating idle mode, it was assumed that the first orbit maneuver was performed in idle mode only, using residual propellants from the main LH_2 tank and the LO_2 and LH_2 lines. For all approaches, main LO_2 tank residual was transferred to the LO_2 maneuver tank after the propellant level dropped past the tank outlet into the supply line (approximately 13 seconds before cutoff at injection).

For both the 10 percent and 20 percent thrust assumptions, burn times are sufficiently short that the incremental velocity requirements are very nearly the theoretical values required assuming impulsive (zero time) firings. In idle mode, however, the available thrust results in very low accelerations and long burn times, and losses compared to impulsive increments become significant. Figure 2-28 gives the percent increases in ΔV for low F/W firings compared with impulsive increments. The average F/W ratio for idle mode is about 0.01, resulting in a 5 percent increase in required ΔV for a 500 ft/sec velocity increment.

Performance for the four approaches is compared in Tables 2-10 and 2-11. Table 2-10 gives the propellants used by the RCS, idle mode, or throttled mode, whichever is applicable, and the boiloff for all phases of the mission. Table 2-11 is an overall comparison of all approaches, omitting weights for components common to all. Idle mode/10 percent gives the lowest system weight, with the 20 percent thrust/no idle mode approach next with 1543 pounds more weight. The combination of lower I_{sp}, higher heat input, and ΔV losses for the pure idle mode approaches make them heavier

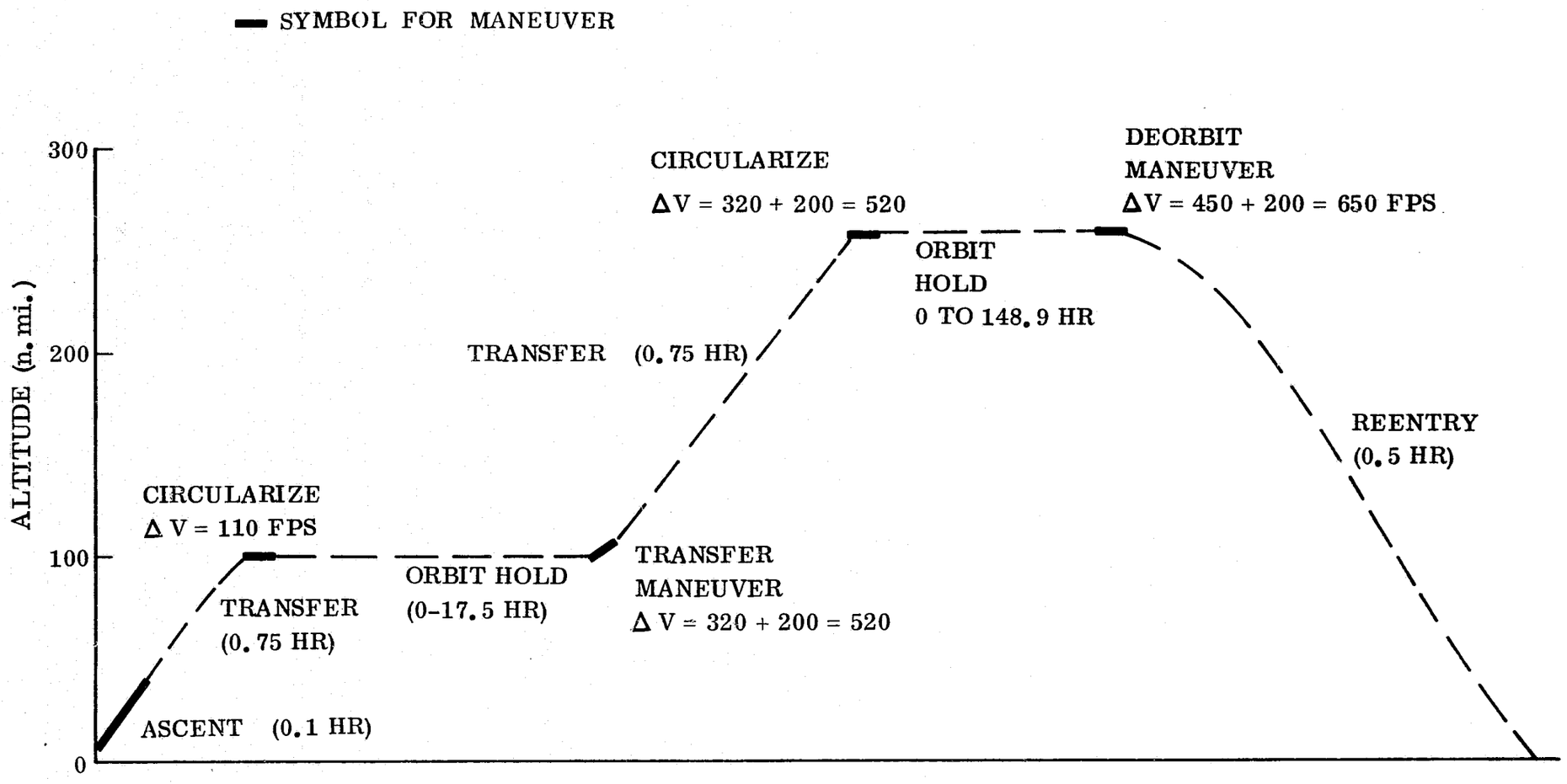


Figure 2-26. Typical Mission Profile

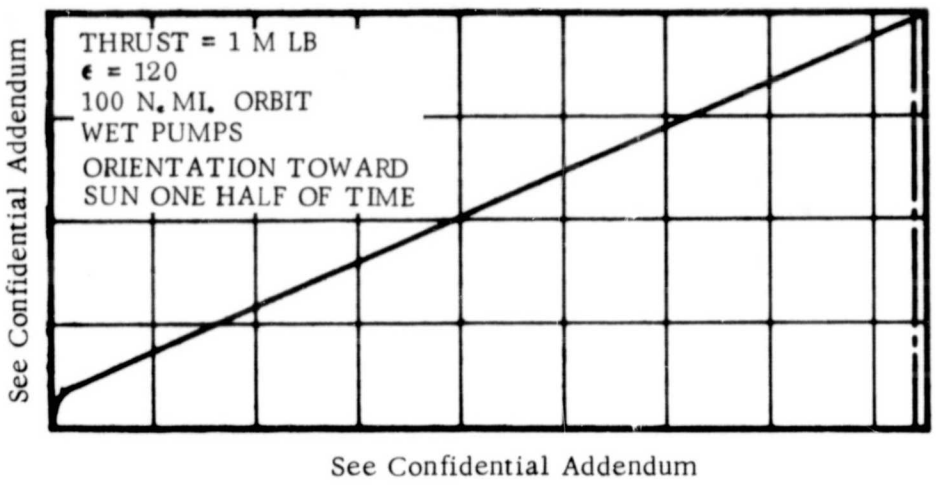
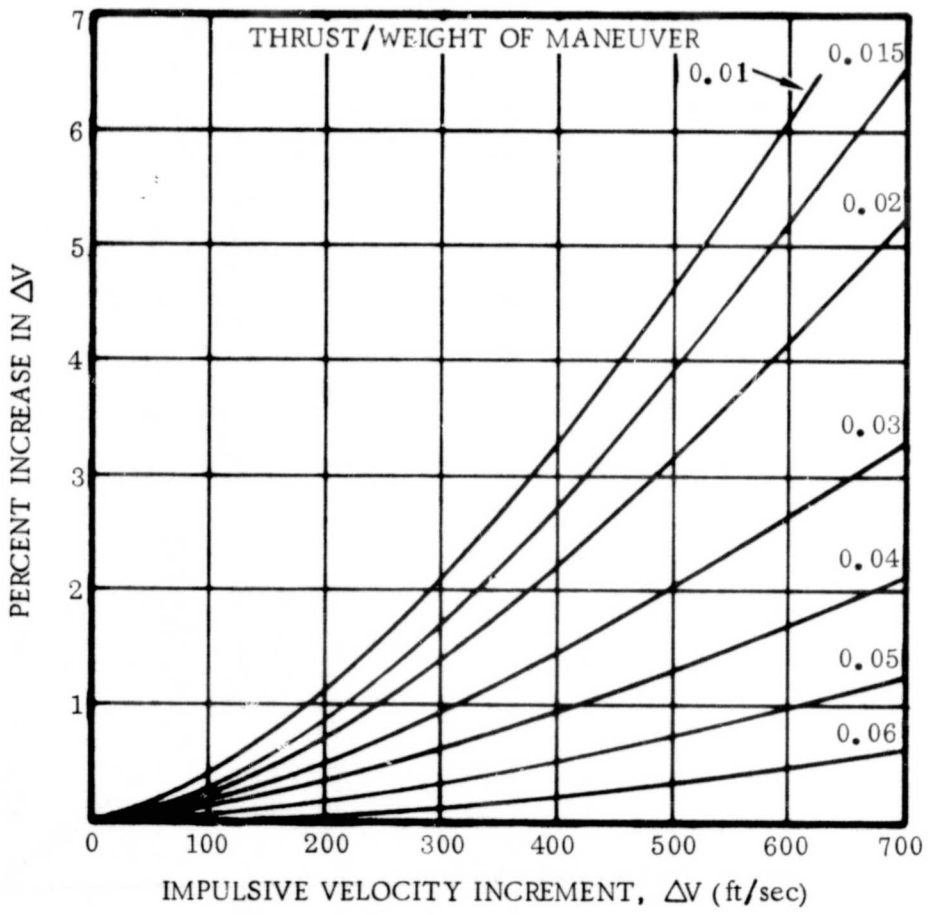


Figure 2-27. Boiloff Losses From High P_C Bell Engine



Figures 2-28. Losses for Low-Acceleration Orbital Maneuvers

Table 2-10. Orbit Maneuver Propellant Usage

	Operation Mode								
	Engine 20% & RCS (1)			Idle Mode Vented (2)		10% & Idle Mode (3)			Idle Mode Unvented (4)
	Boiloff	RCS	20%	Boiloff	Idle Mode	Boiloff	Idle Mode	10%	Idle Mode
Transfer Phase from 45 n. mi. to 100 n. mi.	Tank Pressure Increases From 16 psia at Launch to 30 psia.								
Circularize at 100 n. mi. ΔV = 110 fps		1400 (35 fps)	1976 (75 fps)		3490 (5) (111 fps)(7)		3490(5) (111 fps)(7)		3490(5) (111 fps)(7)
Orbit hold for 17.5 hours	1325			2650		2650			
Transfer maneuver from 100 n. mi. to 260 n. mi. ΔV = 520 fps(6)		1400 (36 fps)	12467 (484 fps)		16750 (546 fps)(7)		133 (4 fps)	13450 (516 fps)	16750 (546 fps)(7)
Transfer Phase from 100 n. mi. to 260 n. mi.	550			110		550			
Circularize at 260 n. mi ΔV = 520 fps(6)		1400 (37 fps)	12015 (483 fps)		16750 (546 fps)(7)		133 (4 fps)	12870 (516 fps)	16750 (546 fps)(7)
Orbit hold for 148.9 hours	7164			14328		7164			
Deorbit ΔV = 650 fps(6)		1400 (39 fps)	14530 (611 fps)		19650 (695 fps)(7)		133 (4 fps)	15433 (646 fps)	19650 (695 fps)(7)
Total Propellant (each type operation) lb	9039	5600	40988	18078	56640	10364	3889	41753	56640
Total Propellant (each mode) lb	55627			74718		56006			56640

(1) 20% operation with one engine after propellant settling with RCS. Tanks vented (zero g) at 30 psia.
(2) Pressure-fed idle mode with two engines. Tanks vented at 30 psia.
(3) 10% operation with one engine after propellant settling in pressure-fed idle mode with one engine. Tanks vented at 30 psia.
(4) Pressure-fed idle mode with two engines. No venting. GH_2 used until tank pressure decreased to 25 psia in each maneuver.
(5) Initial maneuver in 2-engine idle mode using 1500 lb main line residual propellant and 1500 lb maneuver propellant.
(6) ΔV = 200 fps added to maneuver for plane change or other maneuver.
(7) Additional ΔV above nominal reflects ΔV loss caused by low idle-mode thrust (F/W = 0.01).

Table 2-11. Weight Summary — Orbit Maneuvering Propulsion

	Engine 20% & RCS	Idle Mode Vented	10% & Idle Mode	Idle Mode Unvented
H_2/O_2	40,988	56,640	45,642	56,640
RCS Propellant	5,600	—	—	—
H_2 Boiloff	9,039	18,078	10,364	—
Tank Insulation and Mounting	1,541	3,678	2,438	37,441
O_2 Gas Residual	266	368	297	368
H_2 Gas Residual	619	1,073	701	368
O_2 Recirculation System	111	—	—	—
H_2 Recirculation System	126	—	—	—
RCS System	1,195	—	—	—
Main Lines and Tank LH_2 Residual	1,500	—	—	—
Total	60,985	79,837	59,442	94,817

by 20,395 and 35,375 pounds for the vented and locked up tanks respectively. Pure idle mode vented tank approach is the least complex, requiring neither pressurization nor recirculation systems. Idle mode/10 percent is less complex (excluding the engine) than the XLR-129 approach since it does not require a recirculation system.

2.1.7.3 Wet Pump and Dry Pump Comparisons. Following completion of the study, the RCS was changed from earth-storable to H_2/O_2. Data showing propellant required to cool down the 400,000-lb engine pumps from a dry condition was also made available by P&W. The first approach (engine + RCS) was modified, therefore, to reflect the use of the H_2/O_2 RCS. Two additional approaches were also studied. The fifth approach was a modification of the 10% and idle mode approach; after each firing the propellants in the lines were forced back into the tanks by a combination of acceleration furnished by the RCS and helium pressurization. Prior to each successive 10% firing propellant settling and pump cooldown was provided by idle mode operation. The sixth approach was a modification of the XLR-129 and RCS approach, using RCS and helium to return propellant to the tanks after each firing and using RCS for propellant settling while dumping propellant through the dry pumps prior to each successive firing.

The H_2/O_2 RCS consists of gaseous operating thrusters, high pressure gas accumulator tanks and a compressor combination which can reclaim residual gases from the main and maneuver propellant tanks. Steady state specific impulse of the RCS engines is 440 seconds. They operate at a mixture ratio of 4.0.

In the initial study, the approaches that incorporated idle mode used 1500 pounds of residual propellant in performing the first orbit maneuver. The first approach, which did not have idle mode, was therefore penalized because it could not use the residual propellant. Since this approach uses the RCS, which can now utilize residual propellants, this penalty was removed in comparing weights of the approaches.

For the approaches that maintain the dry pump during coast phases, a study was performed to determine the most efficient method of returning line propellants to the maneuver tanks. Methods considered were: (1) Reverse settling with accelerations provided by the RCS. (2) Reverse settling plus expulsion due to pressurization of the lines by heat soak-back. (3) Reverse settling plus line pressurization with a separate helium system. Methods 1 and 2 required long times and excessive acceleration weight penalties. The method chosen used helium stored at ambient (530° F) temperatures and injected into the lines at different pressures so that expulsion times from the two lines were equal. This required 4.3 seconds and an increase in bottle weight of 99 pounds per transfer. The total helium system for three transfers weighs 322 pounds.

Figures 2-29 and 2-30 show the propellant required to cool down the dry pumps as a function of the cooldown time. Figure 2-31 shows the idle-mode thrust as a function o of engine inlet pressure for mixture ratios of 2 and 3. Interpolating the three curves resulted in idle-mode operation at a mixture ratio of 2.4 for 130 seconds to provide

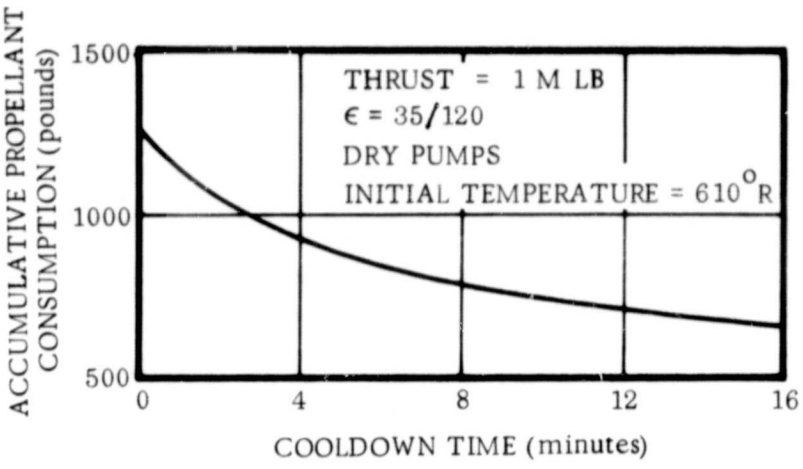


Figure 2-29. Oxidizer Pump Cooldown Propellant Consumption

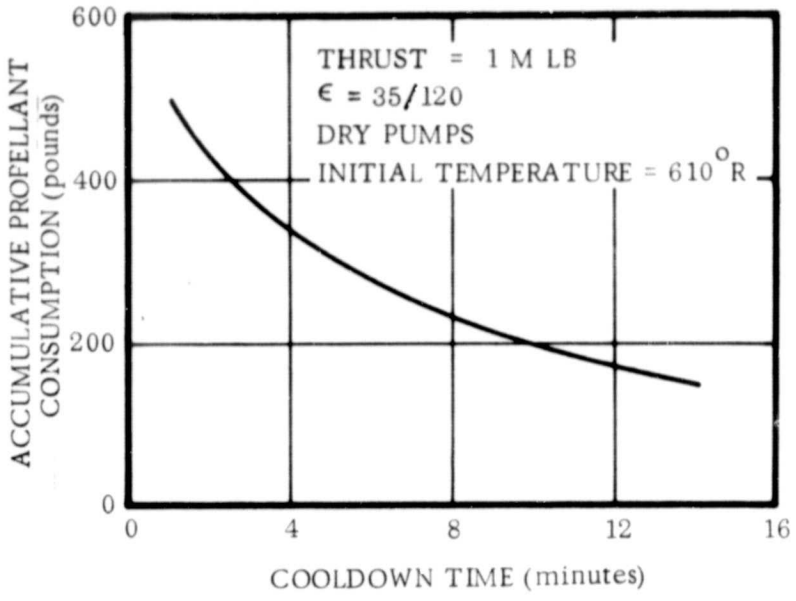


Figure 2-30. Fuel Pump Cooldown Propellant Consumption

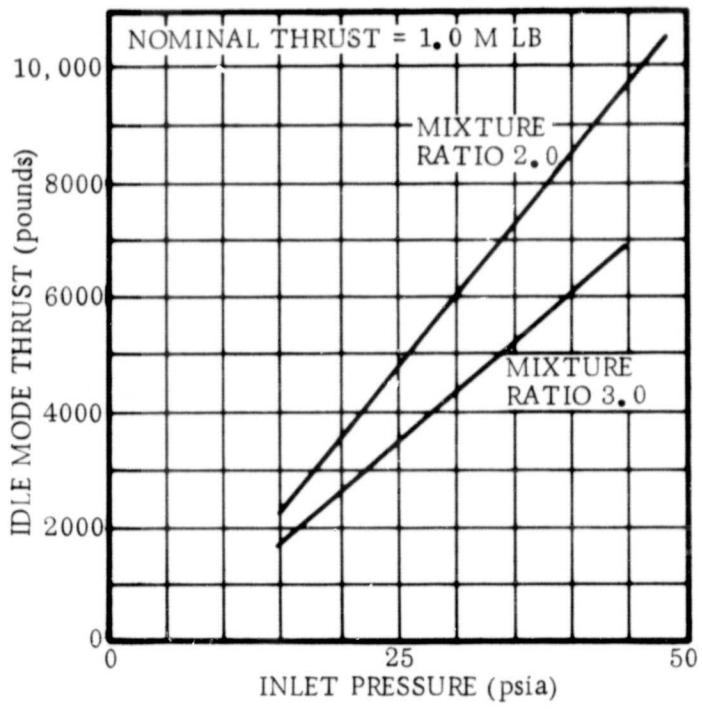


Figure 2-31. Idle Mode Thrust at MR of 2.0 and 3.0

(U) sufficient propellant to cool down the dry pumps. In the 10% and idle-mode approach the main engine operates in the idle mode, using the cooldown propellant for propellant settling. In the engine and RCS approach, idle-mode operation is not available, so cooldown propellant is dumped overboard while propellant settling acceleration is provided by the RCS engine.

(U) Performance, based on the O_2/H_2 RCS and the latest cooldown information, is compared for all six approaches in Tables 2-12 and 2-13. Table 2-12 shows the propellants used in the various modes of operation by each approach for all phases of the mission. Table 2-13 compares the overall weight of the approaches, omitting weights of components common to all. The weights of approaches 2, 3, and 4 have not changed from those shown in Tables 2-10 and 2-11. The lightest is the dry pump 10% and idle mode approach; it is 2,969 pounds lighter than the dry pump engine 20% and RCS approach, and 8,170 pounds lighter than the wet pump engine 20% and RCS approach.

2.1.7.4 Summary. Results of this study show distinct performance improvements for use of the dry pump mode of operation and for the use of a pressure-fed idle mode

for propellant settling and engine chilldown. Use of the dry pump approach requires no new technology and is recommended. Incorporation of pressure-fed idle mode is considered by the engine manufacturers to be a significant design risk that can increase program costs by as much as 25 percent and extend development time by two years. Because of this, it is not recommended for initial incorporation into the space shuttle program. However, once a vehicle design is established, incorporation of idle mode could significantly improve payload or mission flexibility. It is therefore recommended that study of its use continue for eventual incorporation into the program.

2.1.8 NET POSITIVE SUCTION HEAD (NPSH) TRADEOFFS. A tradeoff study was made to determine effects of design NPSH requirements on space shuttle stage weight. The study was made on an FR-4 three-element sequential burn vehicle with a propellant subsystem configuration as shown in Figure 2-32. Factors affecting the NPSH tradeoff, indicated in Figure 3-33, are as follows:

a. Engine Weight. As the engine design NPSH requirement is decreased, the diameter and weight of the low-speed inducer increase. Data on inducer weight and inlet diameter as a function of design NPSH are given in Figure 2-33.

b. Inlet Duct Size. The pressure-volume-compensated (PVC) duct used to provide motion between the engine and the vehicle during engine gimballing also increases in size, along with the inducer, with a decrease in design NPSH requirement. Therefore, both hardware dry weight and liquid residual weight increase. (Ducting diameter above the PVC duct remains constant for this study.)

c. LO_2 Tank Pressure. Two approaches for providing the required LO_2 NPSH have been considered. Because of the LO_2 line length, NPSH can be provided by hydrostatic head alone and the tank pressure is not a function of NPSH required. Height of liquid in the line and the "NPSH residual" requirement then varies with design NPSH. Alternatively, the tank pressure can be varied with design NPSH requirement to eliminate (or reduce to a practical minimum) the NPSH residual, trading tank and pressurant weight against residual. LO_2 tank weights as a function of tank pressure are shown in Figure 2-33. Thrust is reduced to 20 percent as the LO_2 line empties, reducing NPSH requirement and the height of liquid required at engine cutoff for both cases.

d. LH_2 Tank Pressure. Tank pressure is required to supply the LH_2 NPSH, since little hydrostatic head is present. For a decreased LH_2 NPSH requirement, the LH_2 tank pressure and weight decrease on the orbiter and remain the same on the booster, as shown in Figure 2-33.

The explanation for the crossover of the booster and orbiter LH_2 tank weight curves (Figure 2-33) is that the net compressive loads on the booster tank are approximately 50 to 60 percent larger than those on the orbiter tank. This makes a larger percentage of the booster tank weight dependent on compressive loads.

Mission Phase	Wet Pump LR129 20% & RCS (1)			Wet Pump Idle Mode Vented (2)		Wet 10% & Id	
	Boiloff	RCS	20%	Boiloff	Idle Mode	Boiloff	Idle
Transfer Phase from 45 n. mi. to 100 n. mi.	Tank Pressure Increases From 16 psia at Launch to 30 psia.						
Circularize at 100 n. mi. ΔV = 110 fps		986 (35 fps)	1976 (75 fps)		3490 (111 fps)(8)		349 (111
Orbit hold for 17.5 hours	1325			2650		2650	
Transfer maneuver from 100 n. mi. to 260 n. mi. ΔV = 520 fps(7)		986 (36 fps)	12467 (484 fps)		16750 (546 fps(8)		133 (4 f
Transfer phase from 100 n. mi. to 260 n. mi.	550			1100		550	
Circularize at 260 n. mi. ΔV = 520 fps(7)		986 (37 fps)	12015 (483 fps)		16750 (546 fps)(8)		133 (4 f
Orbit hold for 148.9 hours	7164			14328		7164	
Deorbit ΔV = 650 fps(7)		986 (39 fps)	14530 (611 fps)		19650 (695 fps)(8)		133 (4 f
Total Propellant (each type operation) lb	9039	3944	40988	18078	56640	10364	388
Total Propellant (each mode) lb	53971			74718		5	

(U)

(1) 20% operation with one engine after propellant settling with RCS. Tanks vented (zero g) at 30 psia. Propell
(2) Pressure-fed idle mode with two engines. Tanks vented at 30 psia. Propellant remains in lines and pumps.
(3) 10% operation with one engine after propellant settling in pressure-fed idle mode with one engine. Tanks ve
(4) Pressure-fed idle mode with two engines. No venting. GH_2 used until tank pressure decreased to 25 psia i
(5) 10% operation with one engine after propellant settling in pressure-fed idle mode with one engine. Propella
(6) 20% operation with one engine after propellant settling with RCS and pump cooldown. Propellant return to ta
(7) ΔV = 200 fps added to maneuver for plane change or other maneuver.
(8) Additional ΔV above nominal reflects ΔV loss caused by low idle-mode thrust (F/W = 0.01).

FOLDOUT FRAME 1

Table 2-12. Orbit Maneuver Propellant Usage

Operation Mode								
t Pump dle Mode (3)		Wet Pump Idle Mode Unvented (4)	Dry Pump 10% & Idle Mode (5)			LR129 20% & RCS (6)		
e Mode	10%	Idle Mode	RCS	Idle Mode	10%	RCS	Cool Down	20%
0 fps)(8)		3490 (111 fps)(8)		3490 (111 fps)(8)		2067		975 (37fps)
			72			72		
ps)	13450 (516 fps)	16750 (546 fps)(8)		1469 (44 fps)	12407 (476 fps)	2067 (76 fps)	1469	11437 (444 fps)
			72			72		
ps)	12870 (516 fps)	16750 (546 fps)(8)		1469 (45 fps)	11847 (475 fps)	2067 (79 fps)	1469	10970 (441 fps)
			72			72		
ps)	15433 (646 fps)	19650 (695 fps)(8)		1469 (46 fps)	14430 (604 fps)	2067 (82 fps)	1469	13507 (568 fps)
9	41753	56640	216	7897	38684	8484	4407	36889
6006		56640	46797			49780		

ant remains in lines and pumps.

nted at 30 psia. Propellant remains in lines and pumps.
n each maneuver. Propellant remains in lines and pumps.
nt returned to tanks after each firing by reverse acceleration.
nks after each firing by reverse acceleration using RCS plus line pressurization with separate helium subsystem.

Table 2-13. Weight Summary — Orbit Maneuvering Propulsion

	Engine 20% & RCS Wet pump	Idle Mode Vented Wet Pump	10% & Idle Mode Wet Pump	Idle Mode Unvented Wet Pump	10% & Idle Mode Dry Pump	Engine 20% & RCS Dry Pump
H_2/O_2	40,988	56,640	45,642	56,640	46,581	41,296
RCS Propellant	3,944	—	—	—	216	8,484
H_2 Boiloff	9,039	18,078	10,364	—	—	—
Tank Insulation and Mounting	2,225	3,678	2,438	37,441	1,515	1,312
O_2 Gas Residual	266	368	297	368	295	264
H_2 Gas Residual	619	1,073	701	368	342	290
O_2 Recirculation System	111	—	—	—	—	—
H_2 Recirculation System	126	—	—	—	—	—
RCS System	130	—	—	—	7	279
Helium Subsystem	—	—	—	—	322	322
Total	57,448	79,837	59,442	94,817	49,278	52,247

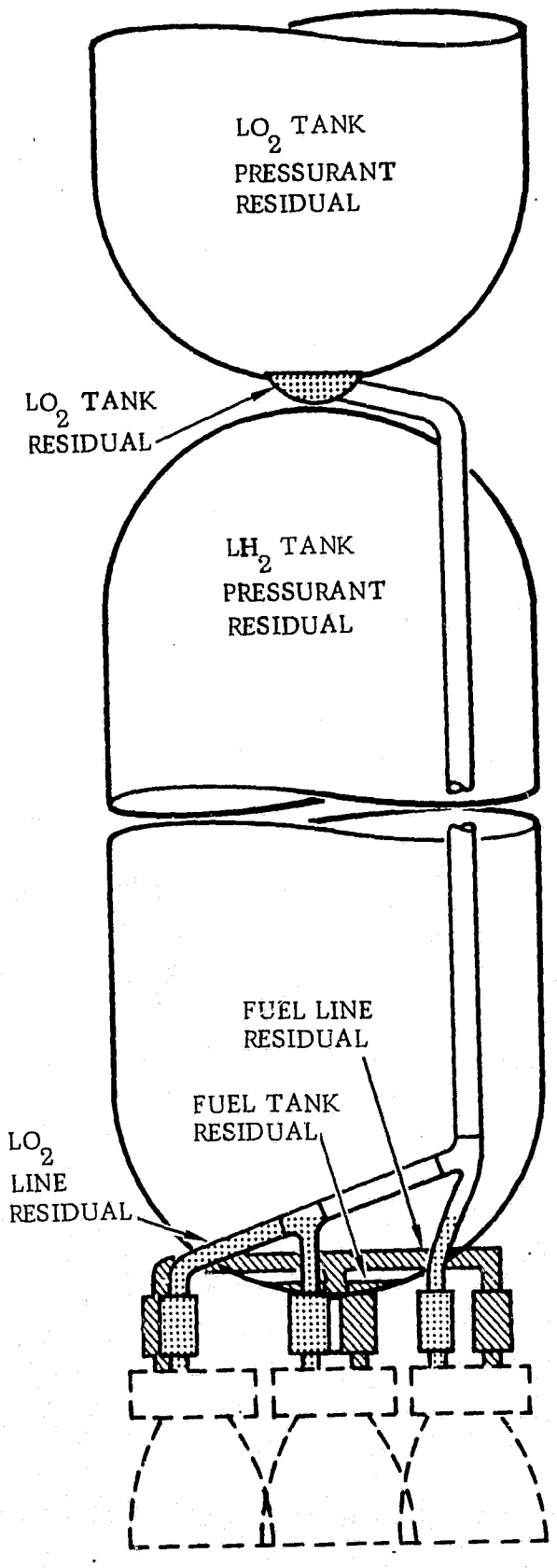


Figure 2-32. Configuration Factors Affecting NPSH Tradeoff and Residuals

The lower pressures reduce the orbiter tank weight but not the booster tank weight for pressure below 30 psia. No pressures above this value were considered in this study.

For LH_2, the nominal-flow NPSH requirement (60 ft) must be used in the tradeoff, since any reduced NPSH requirement due to throttling is achieved only over a period of a few seconds for minimizing tank liquid residuals at depletion (cutoff). Line and tank liquid residuals remain constant.

Figure 2-34 summarizes the NPSH tradeoff for both assumed approaches for the LO_2 subsystem. For the constant-tank-pressure approach, minimum subsystem weight is obtained at the 16-ft NPSH requirement for the orbiter and at 20 ft for the booster. The difference in weight between the 16 and the 20-ft values for the booster is negligible. For the variable-tank-pressure assumption, subsystem weight has not reached bottom at the highest NPSH value considered (32 ft) for either orbiter or booster. Comparing the two approaches, use of variable tank pressure (NPSH residual eliminated) gives the lowest system weights for both orbiter and booster. Differences are fairly minor, however, and pressurizing to use all residual from the LO_2 lines would reduce weight by only 400 lb in the orbiter and 1,000 lb in each booster. In practice, all of the line residuals could not be used, reducing or eliminating these differences.

Figure 2-34 also summarizes the tradeoff for the LH_2 system. Orbiter system weight is minimized at a 90-ft NPSH requirement while the design value is 60 ft, but the reduction at 90 ft is negligible. The booster system weight had not reached bottom at the highest value considered (130 ft). This gave a system weight about 400 lb below the design value.

This tradeoff indicates that there would be no incentive to reduce design NPSH values below the current design values. There is potential for a minor weight reduction at higher design NPSH values, but whether this can actually be achieved would be a function of the detail design of the final vehicle.

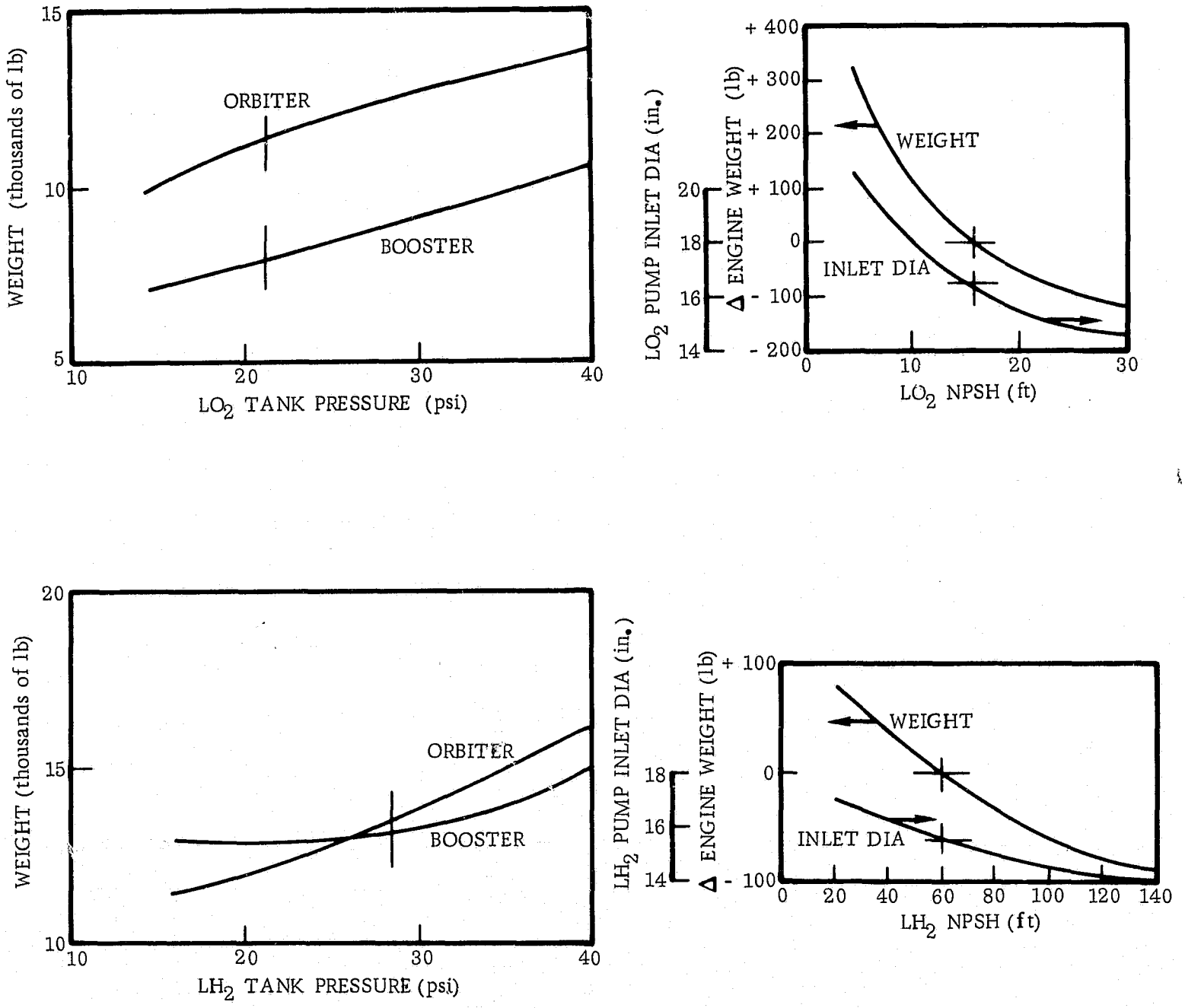
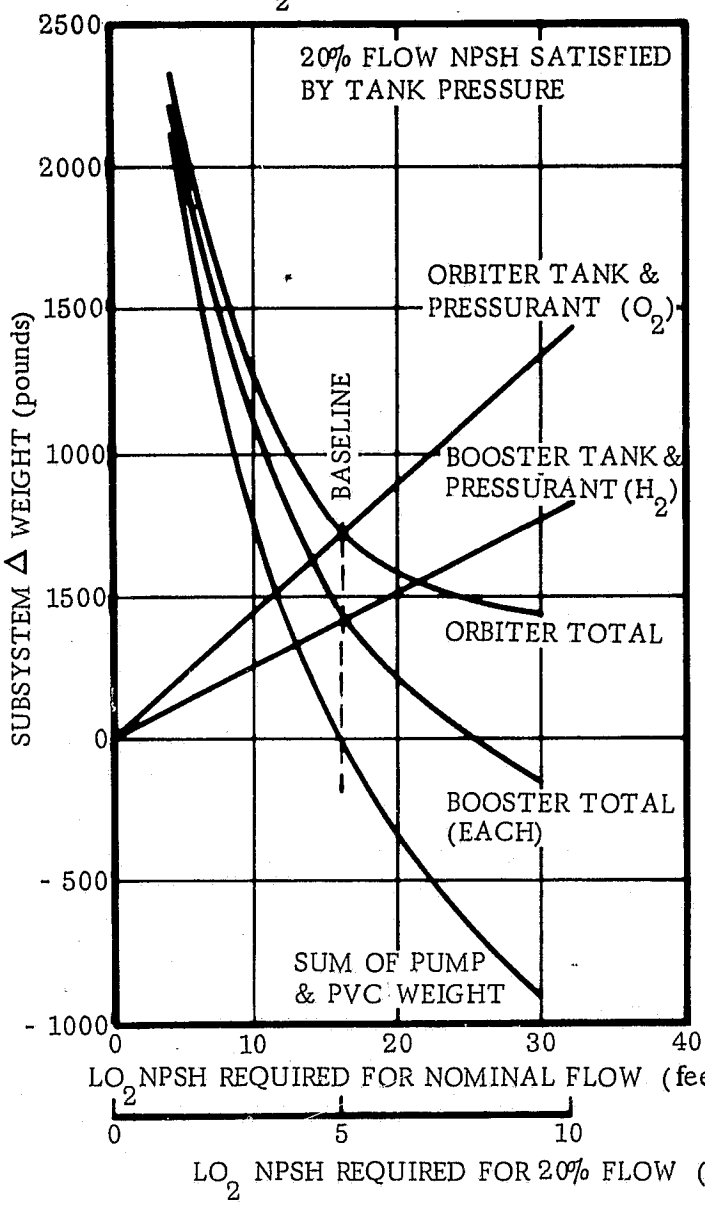
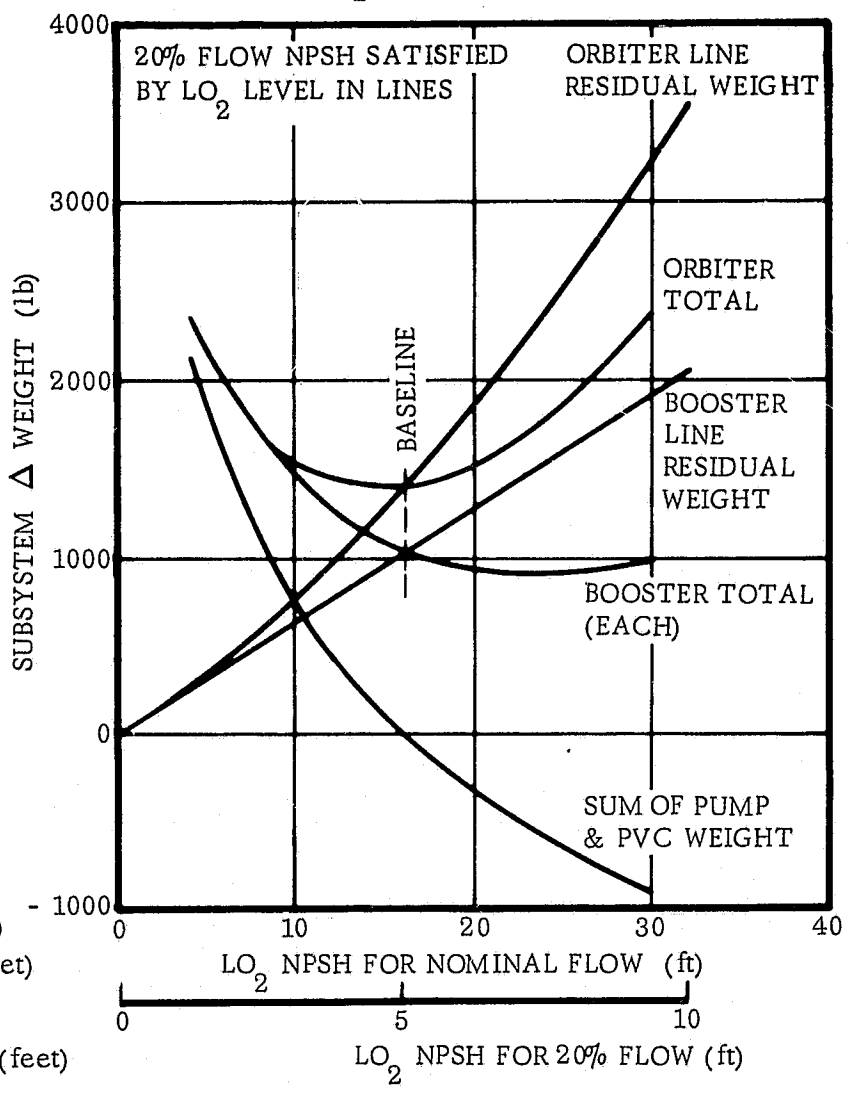


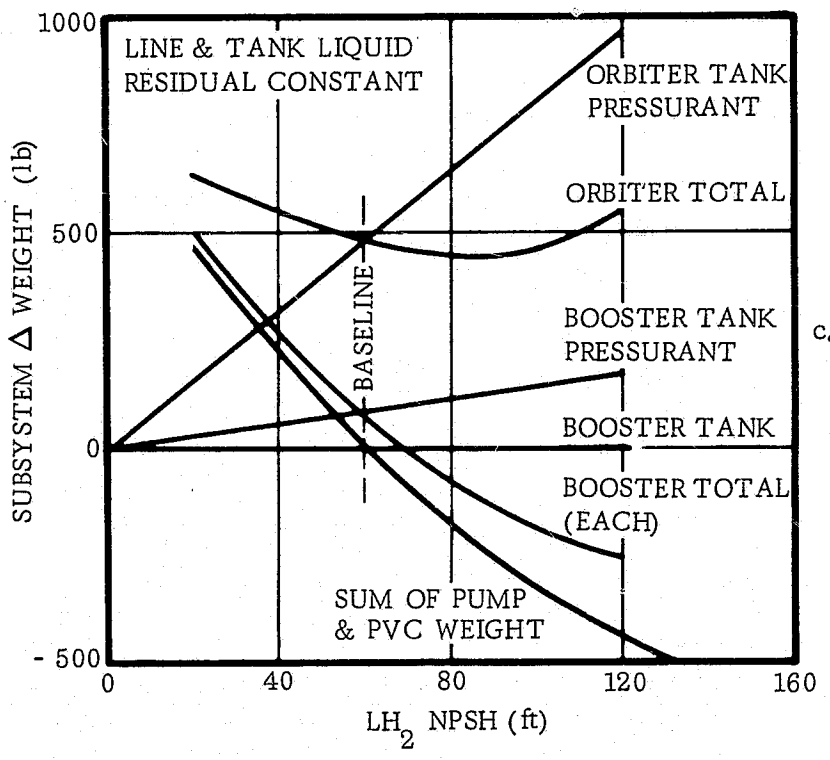
Figure 2-33. Engine and Tank Weight Sensitivity to NPSH and Tank (Ullage) Pressure

a. LO_2 Tank Pressure Variable



b. LO_2 Tank Pressure Constant





c. LH_2 Tank Pressure Variable

Figure 2-34. Propellant Subsystem Sensitivity to NPSH

2.2 EFFECT OF ENGINE SIZE AND NUMBERS ON VEHICLE DESIGN, PERFORMANCE AND COST

2.2.1 INTRODUCTION. The purpose of this study was to examine the effects of incorporating bell engines of three different fixed thrust sizes into a fully reusable logistics vehicle. Design and development factors involved in incorporating engines whose thrust values at sea level are 300,000 pounds, 400,000 pounds, and 500,000 pounds were examined. All factors of development and operations were considered, leading to a total cost analysis.

The following sections define the groundrules applied to the study analysis, design requirements derived, the synthesis of candidate vehicle systems, design of selected vehicles, and specific technical data supporting selection of vehicles on which to base costs, and results of the cost analysis.

Fully reusable logistics vehicles studied are the parallel burn, three-element FR-1 vehicle system and the two-element sequential burn FR-3 vehicle system. Both of these vehicles have the capability of transporting a 50,000-pound payload, 15-ft diameter × 60-ft long to a low earth orbit and returning the same or a similar payload to earth. A 25,000 pound payload FR-1 system in a 15-ft diameter × 30-ft long bay is also included.

The design studies concentrated on major functional considerations and requirements that define the aft body structure, engine mounting, and engine gimballing functional clearances. Prime importance was given to the evolution of fuel supply ducting, distribution manifolds, and articulated flexible ducting to provide for engine functions and structural deflections.

2.2.2 GROUNDRULES AND REQUIREMENTS

2.2.2.1 Groundrules. The following groundrules form the basis for the multi-engine derived requirements and subsequent design studies:

a. Vehicle basic configuration:
   1. FR-1A 50K lb payload configuration T-18N.
   2. Two-element, sequential burn 50K lb payload configuration M-2A (FR-3).

b. Costs to be based on mission/payload logistics for 50 launches/year and 100 launches/year.

c. Size engine for 300-400-500K pound thrust at sea level.

d. Commonality — same basic engine for booster and orbiter.

e. Engine to be bell with two-position nozzle.

f. Provide engine gimballing fuction with one engine set in null position.

g. Provide gimballing function to satisfy one-engine-out condition.

h. All hydrogen supply plumbing to be jacketed.

i. All bellows allowing engine gimbal excursions to be safety jacketed pressure-volume-compensated (PVC) type.

2.2.2.2 Mission Requirements

Orbit altitude (circular)	270 n.mi.
Inclination	55 deg.
Launch facility	ETR
Orbit duration	7 days

2.2.2.3 Design Requirements

Axial load limit (performance)	3g
Axial load limit (structure)	4g
Factor of safety (general)	1.4
Burst factor (tanks)	1.5
Weight contingency on vehicle dry weights	10%
Engine thrust effectiveness (liftoff/nominal)	114%/100%
Fuel/oxidizer mixture ratio (liftoff/nominal)	6.4/7.0

Flexible engine inlet duct — pressure volume compensated type

50K lb payload, 15 foot diameter × 60 ft long

2.2.3 OVERALL VEHICLE CONFIGURATIONS. Vehicles used as the point of departure for this study are defined in Figures 2-35 and 2-36. The two-element vehicle has 8 booster engines and 2 orbiter engines. Nominal sea level thrust is 602,000 pounds. The orbiter contains the 60-foot long by 15-foot diameter payload bay. The orbiter has a Vee tail for stability across hypersonic, transonic, and subsonic velocity range. It has a flat bottom for good hypersonic L/D and a full-radius upper surface to accommodate cylindrical propellant tanks as shown. The wing used subsonically stows below the hydrogen tank and payload bay. The booster is similar in shape except that a horizontal tail is added below the Vee tail for stability across the flight regime with the larger weight of rocket engines. The nominal cg of the orbiter is at 55 percent of the body length and of the booster is at 60 percent of the body length.

All the vehicles have blunt bases since the increase in performance due to the possible increase in rocket engine expansion ratio and specific impulse far outweigh the decrease in booster flyback L/D in a boattailed afterbody configuration.

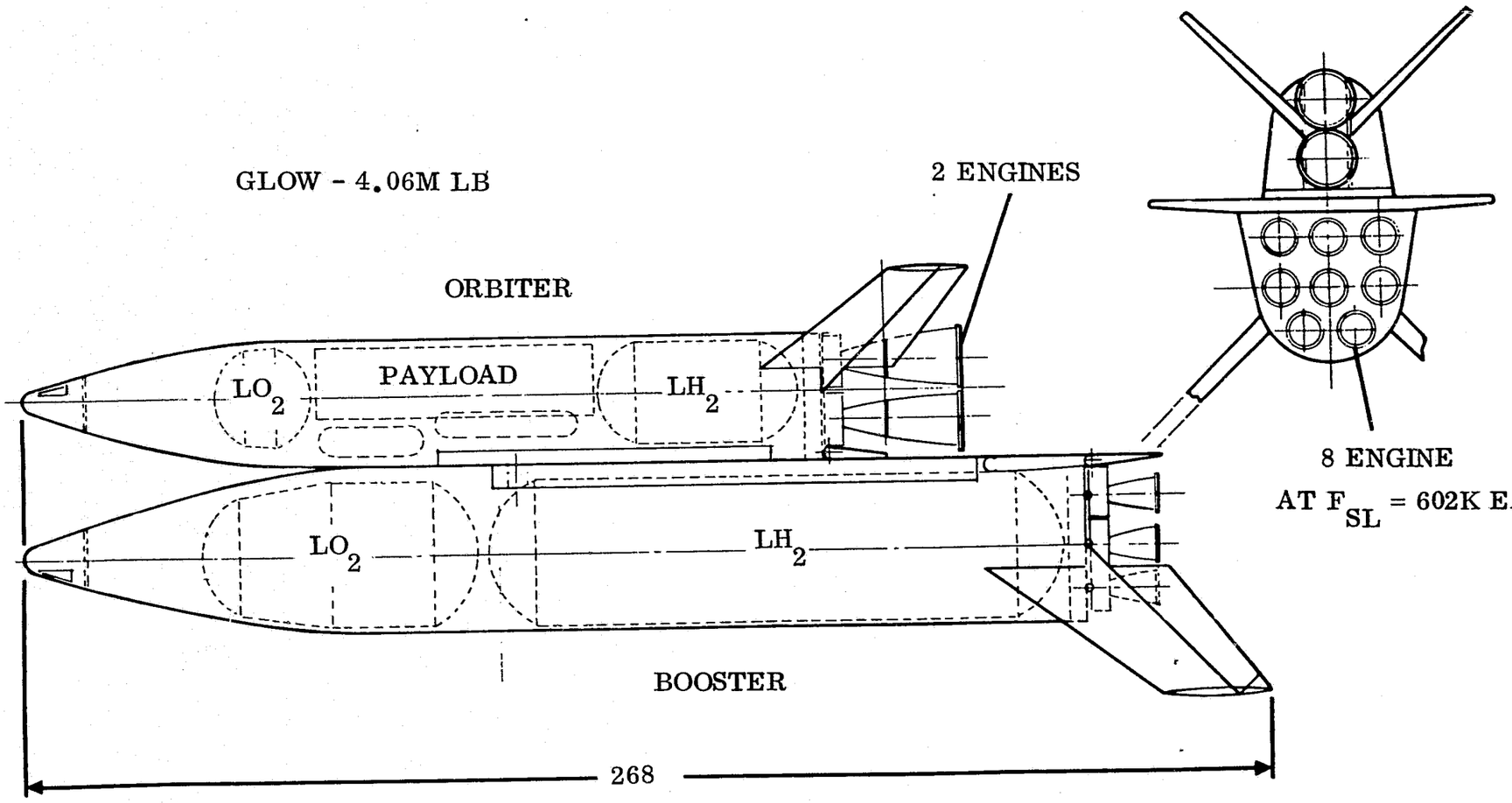


Figure 2-35. Baseline Two-Stage System — General Arrangement, Sequential Burn, 50,000 Lb Payload

Figure 2-36. Baseline FR-1 Element Three-View and Launch Arrangement (2-2-2 Engine Arrangement)

The booster has eight rocket engines and the orbiter has two. The rocket engines have common thrust chambers and pumps and are the same except for the added nozzle extension and expansion ratio used in the orbiter element.

Element shapes for the three-element parallel burn FR-1 vehicle are similar to those described for the two-element system. All elements are identical in dimensions. The launch configuration of the orbiter (between the two booster elements) is shown in Figure 2-36. All configurations have Vee tails and lower-surface elevons for hypersonic roll control. Two rocket engines are used in each element. These two-position nozzles have expansion ratios of 35/115. The arrangement of two engines per element is shown symbolically as 2-2-2. The individual engine sea level nominal thrust is 984,000 pounds or almost 1,000K pounds.

2.2.4 MULTI-ENGINE VEHICLE CONFIGURATION SELECTION. Initially a number of engine arrangements were investigated for each engine thrust size considered. Vehicles with these engine arrangements were synthesized on the weight/volume/performance synthesis program, with the inputs kept consistent to maintain an objective comparison. All the boosters employed mixture ratios of 6.4, with 7.0 in the orbiters. Payload was fixed at 50,000 pounds and gross liftoff weight was allowed to vary. Thrust-to-weight ratios were outputs, since input thrust was fixed. This results in vehicles of different F/W values, which must be remembered in making comparisons. All vehicles were flown to a staging dynamic pressure of 50 psf with an initial orbit injection point of 43 n.mi. Two-element vehicles were all staged at a nominal 11,000 fps which is near optimum. FR-1 vehicles all have a staging velocity unique to the particular system due to the equal-element criteria. This staging velocity is in the order of 8,200 fps.

All the cases run on the synthesis program are summarized in Tables 2-14 and 2-15. In some case an insufficient number of engines was assumed and the vehicle would not fly. The main features of thrust to weight, GLOW and total system dry weight are shown. Selected vehicles in each category are indicated.

Primary selection criteria were as follows:

a. F/W at liftoff. In general, about 1.45 tends to give optimum performance; above this value the gross structure weights and engine weights increase faster than the gains achieved by reduced gravity losses.

b. Orbiter F/W at staging. This is important since a minimum of approximately 0.825 is currently estimated for a "once around" abort using all available maneuver propellant, with one engine out. The greater the number of orbiter engines the easier this criterion is to meet.

c. Gross liftoff weight.

d. Total system hardware weight.

e. Overall vehicle layout arrangement.

Table 2-14. Multiple Engine Study Survey Results

Vehicle	Thrust/Eng. S.L. 100% Nom. (lb)	Engine Arrang.	$(F/W)_B/(F/W)_O$	GLOW (lb)	Inert Weight (lb)	Selected Cases	Remarks
Two-Stage	500K	10-2	1.426/1.46	4.086M	649K		Low on abort F/W
		10-3	1.36/2.06	4.27M	675K	X	High orbiter F/W
Sequential	400K	12-3	1.23/1.50	4.54M	697K		$(F/W)_B$ low
		13-3	1.46/1.74	4.14M	660K	X	
50K Lb Payload	300K	16-3	Vehicle Would Not Converge				Thrust too low
		16-4	1.28/1.58	4.37M	677K		$(F/W)_B$ low
		17-4	1.45/1.73	4.11M	654K	X	
	602K	8-2	1.392/1.72	4.06M	641K	X	Baseline
FR-1	500K	4-4-4	1.42/1.68	4.88M	824K		$\epsilon = 35/125$
		4-4-4	1.44/1.72	4.81M	817K	X	$\epsilon = 35/140$
50K Lb	400K	5-4-5	1.321/1.35	4.89M	809K		$(F/W)_B$ low
		5-5-5	1.45/1.71	4.79M	807K	X	
Payload	300K	7-5-7	Vehicle Would Not Converge				Thrust too low
		7-6-7	1.41/1.51	4.90M	827K	X	
		7-7-7	1.47/1.74	4.94M	838K		
	984K	2-2-2	1.391/1.63	4.84M	810K	X	Baseline

Table 2-15. Multiple Engine Study Survey Results

Vehicle	Thrust/Eng. S.L. 100% Nom. (lb)	Engine Arrange.	$(F/W)_B/(F/W)_O$	GLOW (lb)	Inert Weight (lb)	Selected Cases	Remarks
FR-1	500K	3-2-3	1.64/1.46	2.81M	499.5K	No	F/W at liftoff too high
25K Lb	400K	3-3-3	1.454/1.722	2.83M	489K	Yes	
Payload	300K	4-4-4	1.39/1.63	2.99M	510K	Yes	
		4-3-4	1.24/1.19	3.07M	510K	No	F/W orbiter too low
	570K	2-2-2	1.391/1.63	2.81M	481K	Yes	Baseline

2.2.5 SELECTED CONFIGURATIONS. The output of Section 2.2.4 culminated in the baseline vehicle systems shown in Table 2-16. The table shows gross characteristics only: engine thrust size, number and combination of engines, and total vehicle weight.

Table 2-16. Effects of Engine Thrust

Vehicle System	Engine Thrust Sea Level (lb $\times 10^3$)	Engine Combination	GLOW (lb $\times 10^6$)
Two-Element Sequential Burn	600	8-2	4.06
	500	10-3	4.27
	400	13-3	4.14
	300	17-4	4.11
FR-1	1000	2-2-2	4.84
	500	4-4-4	4.81
	400	5-5-5	4.79
	300	7-6-7	4.90

This section describes two of these baseline vehicles, representing one configuration from the FR-1 group and one from the FR-3 group. The latter part of the section presents summaries of vehicle characteristics of all the selected baseline vehicles.

2.2.5.1 FR-1 Configurations. Design characteristics of the selected FR-1 multi-engine configurations are summarized in Figure 2-37. A typical aft end for a 50K lb payload vehicle is shown in Figure 3-38. Four bell nozzle engines are gimballed from a system of crossbeams which attach at their outboard ends to the thrust skirt which is an extension of the fuel tank cylindrical section. Internal crossbeams are used to stabilize side loads and to support the gimbal actuators. A dual oxidizer feed is shown, branching out to supply two engines per subsystem. Only the oxidizer lines and PVC ducts are shown, the fuel subsystem manifold being inside the aft bulkhead with four individual penetrations to the respective engines.

The four-engine arrangement, having smaller (500K) thrust engines of shorter length than the 984K two-engine version allows for easier heat shielding by the elevons when the nozzle extension is retracted. While more complex than a two-engine element, the increased complexity would not appear to materially jeopardize accessibility and maintenance.

The arrangement shown in Figure 2-38 includes engines with an expansion ratio of 35/125. The finally selected configuration used an expansion ratio of 35/140 when this was found to be achievable. Data on a 25K lb payload configuration is shown in Figure 2-39 and Table 2-17.

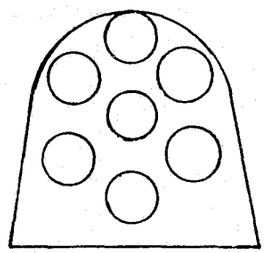
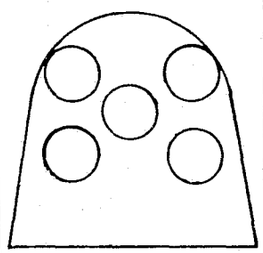
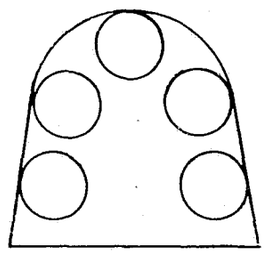
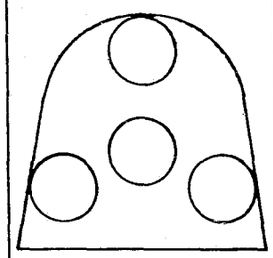
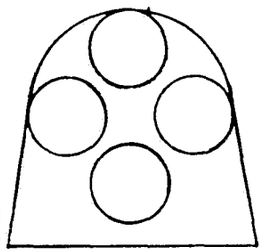
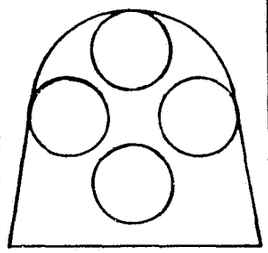
Eng. Size-Nominal	300,000-Lb SL Engines		400,000-Lb SL Engines		500,000-Lb SL Engines	
Engine Data:	Booster	Orbiter	Booster	Orbiter	Booster	Orbiter
No. of Engines	7	5	6	4	4	4
Exp. Ratio	125/35	125/35	125/35	125/35	125/35	125/35
Length (Inch):						
Extended	180	180	210	210	236	236
Retracted	132	132	147	147	165	165
Nozzle Dia. (Inch)	100	100	113	113	129	129
Pr. Pack Dia. (Inch)	80	80	92	92	104	104
Gimbal Angle	5° Up, Down $\pm 3^{\circ}$ Side	$\pm 3^{\circ}$	$\pm 3^{\circ}$	$\pm 3^{\circ}$	$\pm 3^{\circ}$	$\pm 3^{\circ}$
Distance from Gimbal Axis to Tank Blkhd (ft)	4.5	4.5	5.0	5.0	5.5	5.5
Engine Configuration (Number)	 (SEVEN)	 (FIVE)	 (SIX)	 (FOUR)	 (FOUR)	 (FOUR)

Figure 2-37. 50,000-Lb FR-1 Concepts

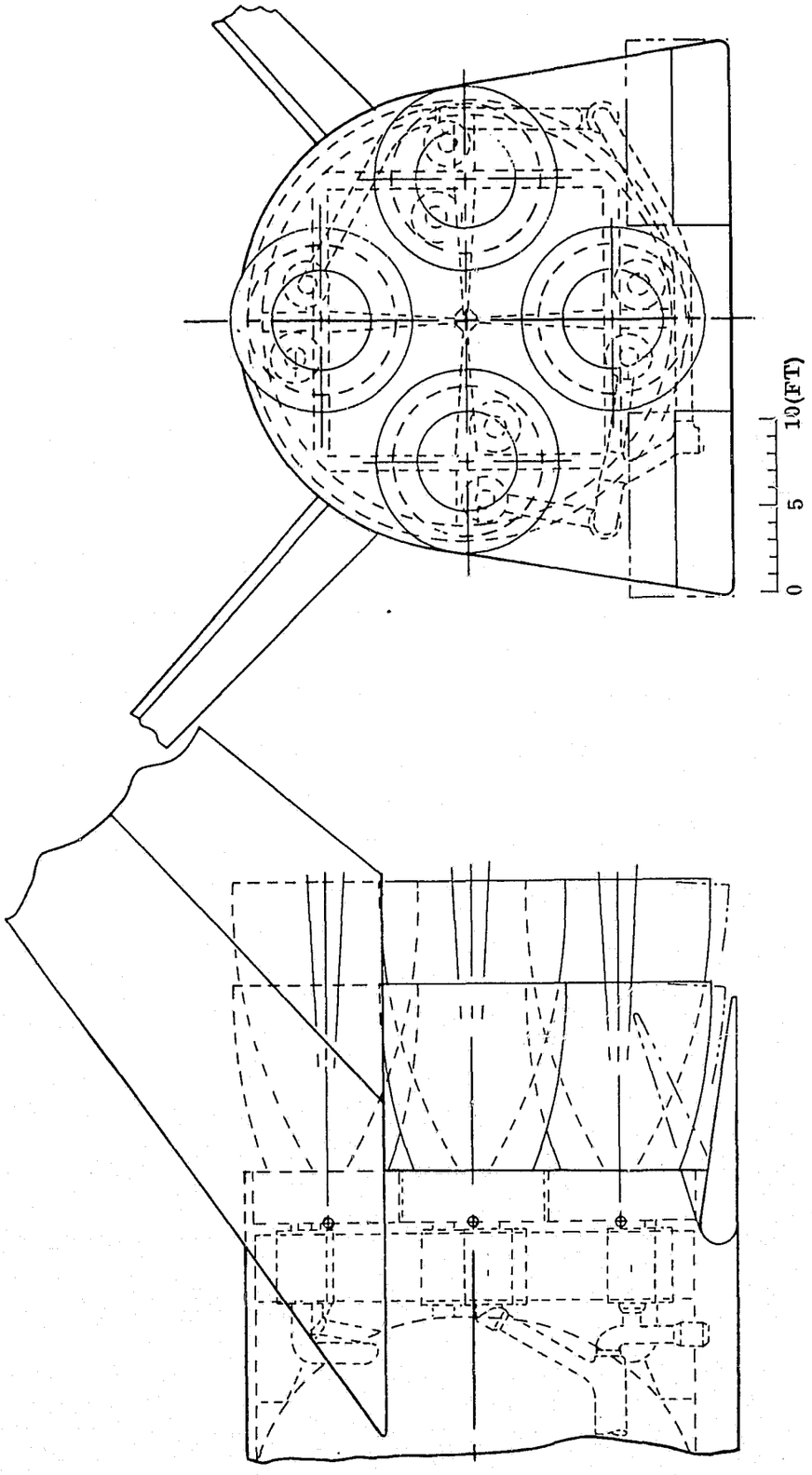


Figure 2-38. 4-4-4 T18N FR-1 500K Eng. ∈125/35 Gimbal Angle ±3°

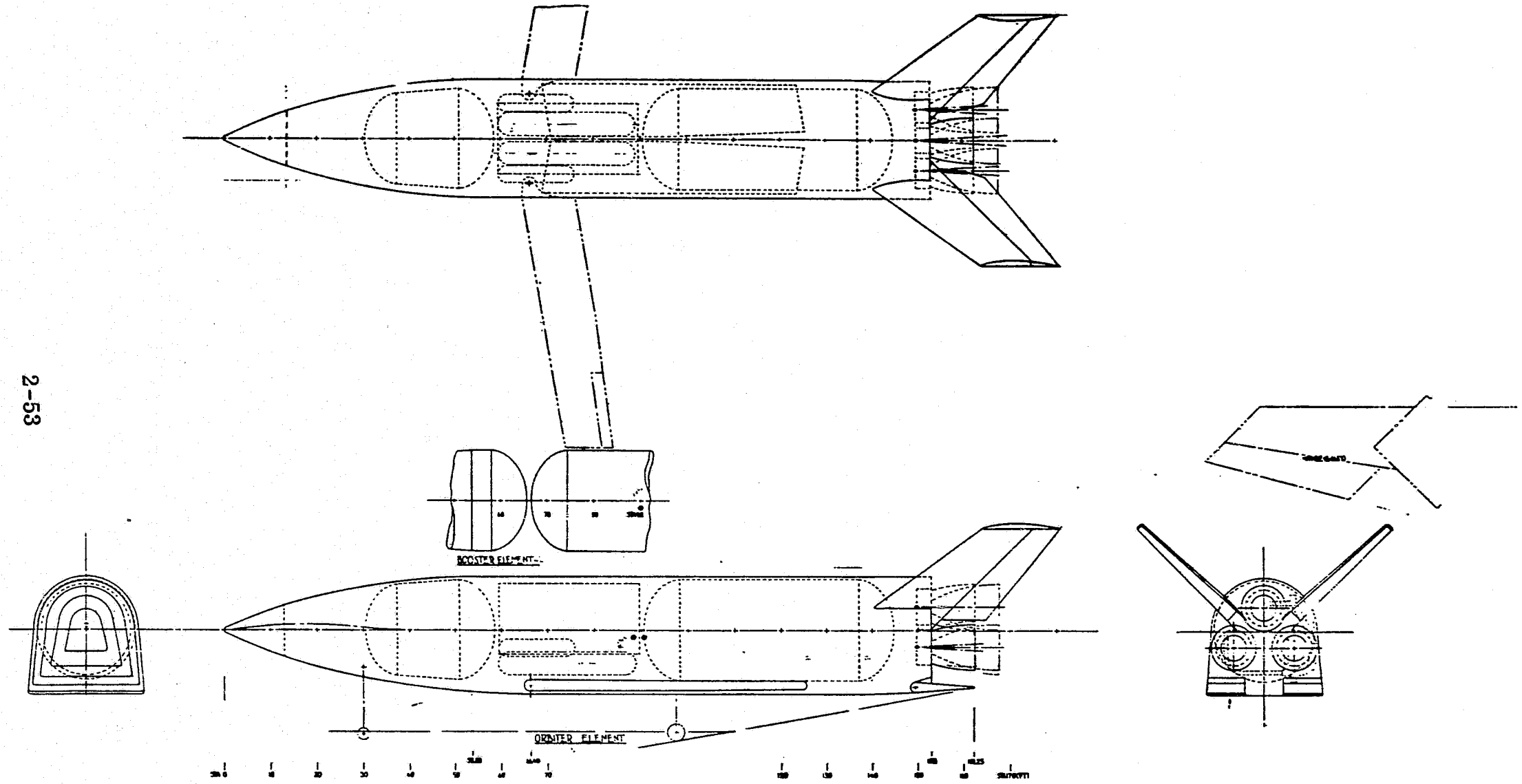


Figure 2-39. Typical Three-Element 25K Lb Payload Vehicle

Table 2-17. Selected Concepts for 25,000 Lb Payload FR-1

	Baseline		Selected Concepts	
Engine Thrust (1000 Lb)	570	500	400	300
Engine Nos. B-O-B	2-2-2		3-3-3	4-4-4
GLOW (Million Lb)	2.81		2.83	2.99
Total Dry Weight (1000 Lb)	481		489	510
ϵ (Booster & Orbiter)		3-2-3 and 2-3-2 Both Unacceptable	See Confidential Addendum	
I_{SP} VAC.				
$(F/W)_B/(F/W)_O$	1.391/1.63		1.454/1.722	1.39/1.63

2.2.5.2 FR-3 Configurations. Design characteristics of the selected FR-3 multi-engine configurations are summarized in Figure 2-40 and a typical aft end for a 50K lb payload vehicle in Figures 2-41 and 2-42.

a. Booster. The configuration selected as the baseline vehicle for the two-element sequential burn boost vehicle, shown in Figure 2-41, consists of the following engine ensemble:

Number of engines	17
Thrust, 100% SL (lb)	300K
Thrust, vac (lb)	348K
Expansion ratio	35/80
Length extended (in.)	140
Length retracted (in.)	88
Nozzle diameter (in.)	78
Power pack dia. (in.)	78

The engine arrangement is in four rows of 4,4,4 and 5 engines from top to bottom. This arrangement resulted from a minimization of fairing areas, maintaining gimbal clearance, and minimum weight engine mounting structure.

This configuration is based on full gimbal angle clearance between engines and provides clearance between engines with one engine in any extreme position and the other engines being gimballed through their full gimbal range. Engines are installed canted up 8 deg to the vehicle centerline. This requirement is due to cg offset of the combined boost/orbiter vehicle. Engine gimbal angles and engine cant requirements were the same for all engine concepts.

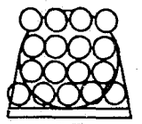
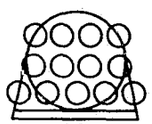
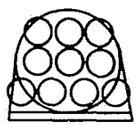
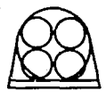
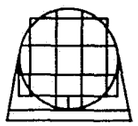
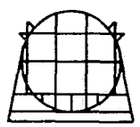
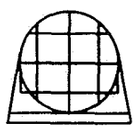
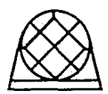
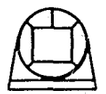
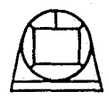
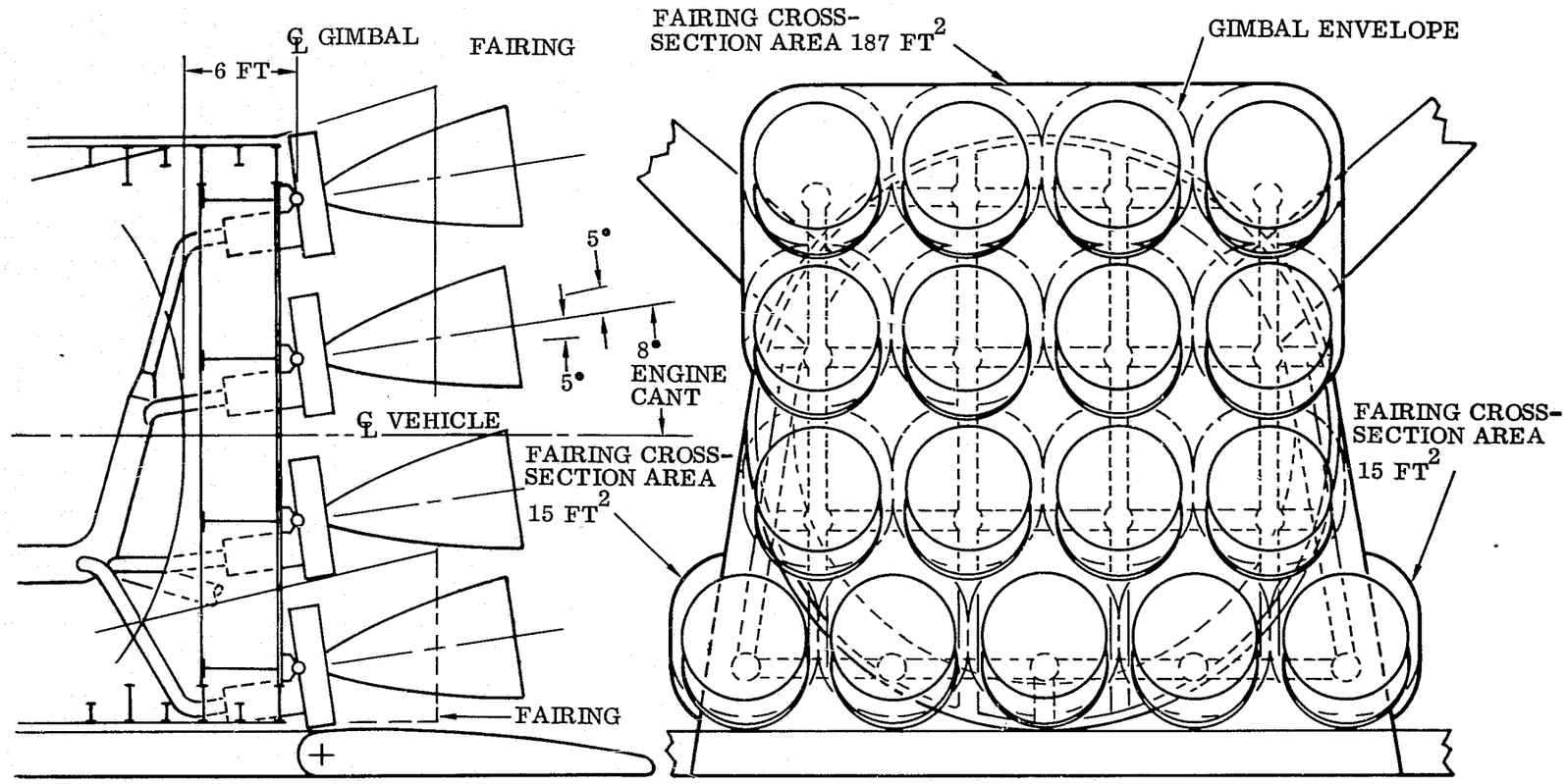
	BOOSTER			ORBITER		
ENGINE SIZE (NOM)	300 K	400 K	500 K	300 K	400 K	500 K
Engine Data:						
Thrust (100% SL)	300,000	400,000	500,000			
Thrust (VAC)	348,000	465,000	581,000	356,000	474,000	594,000
No. of Engines	17	13	10	4	3	3
Expansion Ratio	35/80	35/80	35/80	80/150	80/150	80/150
Extended Length	140	165	185	195	225	260
Retracted Length	88	104	111	114	126	144
Nozzle Dia	78	90	102	108	126	140
Power Pack Dia	78	90	102	78	90	102
Engine Cant Angle	8°	8°	8°	—	—	—
Gimbal Requirements						
Up	5°	5°	5°	3°	3°	3°
Down	5°	5°	5°	3°	3°	3°
Left	3°	3°	3°	3°	3°	3°
Right	3°	3°	3°	3°	3°	3°
Fairing Area (ft^2)						
Null Center	86	144	90	—	—	—
Gimbal Clearance				—	—	—
Distance from Gimbal Axis to Aft Tank Bulkhead (ft)	6.0	6.0	6.0	4.5	5.0	5.5
Engine Arrangement and Number of Engines	 17	 13	 10	 4	 3	 3
Structural Arrangement						

Figure 2-40. Two-Stage Sequential Burn FR-3 Vehicle (50K-Lb Payload)



ENGINE DATA:		ENGINE GIMBAL DATA:
THRUST, SEA LEVEL (LB)	300,000	
THRUST, VAC (LB)	348,000	
EXPANSION RATIO	35/80	
LENGTH EXTENDED (IN)	140	5° UP } FROM 8° CANT LINE
LENGTH RETRACTED (IN)	88	5° DOWN }
NOZZLE DIA (IN)	78	3° EACH SIDE
POWER PACK DIA (IN)	78	FULL ANGLE GIMBAL CLEARANCE BETWEEN ENGINES

Figure 2-41. Two-Stage Sequential Burn Boost Vehicle (50K-Lb Payload, 17-300K-Lb Thrust Engine Ensemble)

ENGINE DATA
THRUST (LB) 356K VAC
ϵ = 80/150
LENGTH (IN.)
RETRACTED 114
EXTENDED 195
NOZZLE DIAMETER (IN.) 108
PWR PK DIA (IN.) 78

Figure 2-42. Two-Stage Sequential Burn Orbiter Vehicle (50K-Lb Payload, Four-300K-Lb Thrust Engine Ensemble)

A distance of six feet is maintained between the aft tank bulkhead and the engine gimbal axis. This distance is governed by structural requirements and the fuel and oxidizer lines and manifold configurations. Fuel manifolding is accomplished inside the fuel tank. Two main oxidizer supply lines are required, one on each side of the tank, manifolded to eight and nine engines.

The engine thrust structure is a multi-beam structure, reacting engine thrust loads. The structural arrangement is determined by engine location, engine gimbal actuator attach points, and engine fuel and oxidizer inlet location. These factors vary with each engine concept, resulting in a different structural arrangement for each concept.

b. Orbiter. The configuration selected as the baseline vehicle for the two-element sequential burn orbiter, shown in Figure 2-42, consists of the following engine ensemble:

Number of engines	4
Thrust, vac (lb)	356K
Expansion ratio	80/150
Length extended (in.)	195
Length retracted (in.)	114
Nozzle diameter (in.)	108
Power pack dia. (in.)	78

Commonality is maintained between the booster and orbiter by using the same basic engine with different expansion ratios.

The engine arrangement is two rows of 2. No fairings are required. Clearance is maintained between the engines, with one engine in the null position and the other engines gimbaling together through the full gimbal range. Engines are installed parallel to the vehicle centerline.

Fuel manifolding is accomplished inside the tank. One main oxidizer supply line is manifolded to the four engines.

2.2.5.3 Selected Vehicle Weight Summary. Weights for the selected multi-engine configurations are summarized in Figures 2-43 and 2-44 in bar chart form.

2.2.6 DEVELOPMENT AND OPERATIONS CONSIDERATIONS. Variations in the space shuttle development and operations programs as a result of engine thrust-level selection will be reflected as delta time and cost factors; neither is considered to have any significant effect on the total program costs. In general, as thrust is increased and number of engines decreased, propulsion and support subsystem hardware components (e.g., propellant distribution lines, valving, and TVC components) are reduced in number. Effects of this hardware reduction on the development and operations programs are briefly described in the following sections.

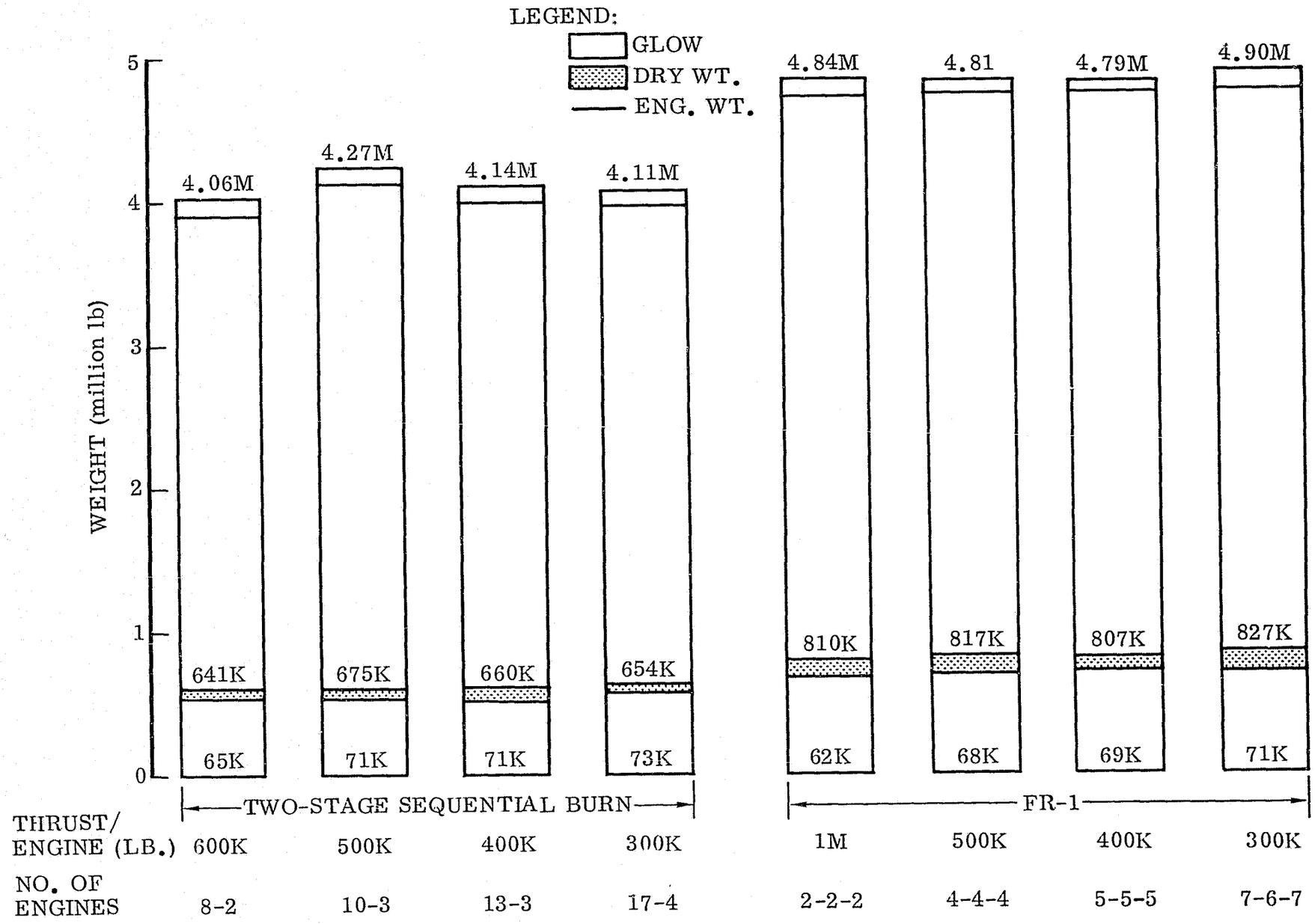


Figure 2-43. Summarized Weights 50K Lb Payload Systems

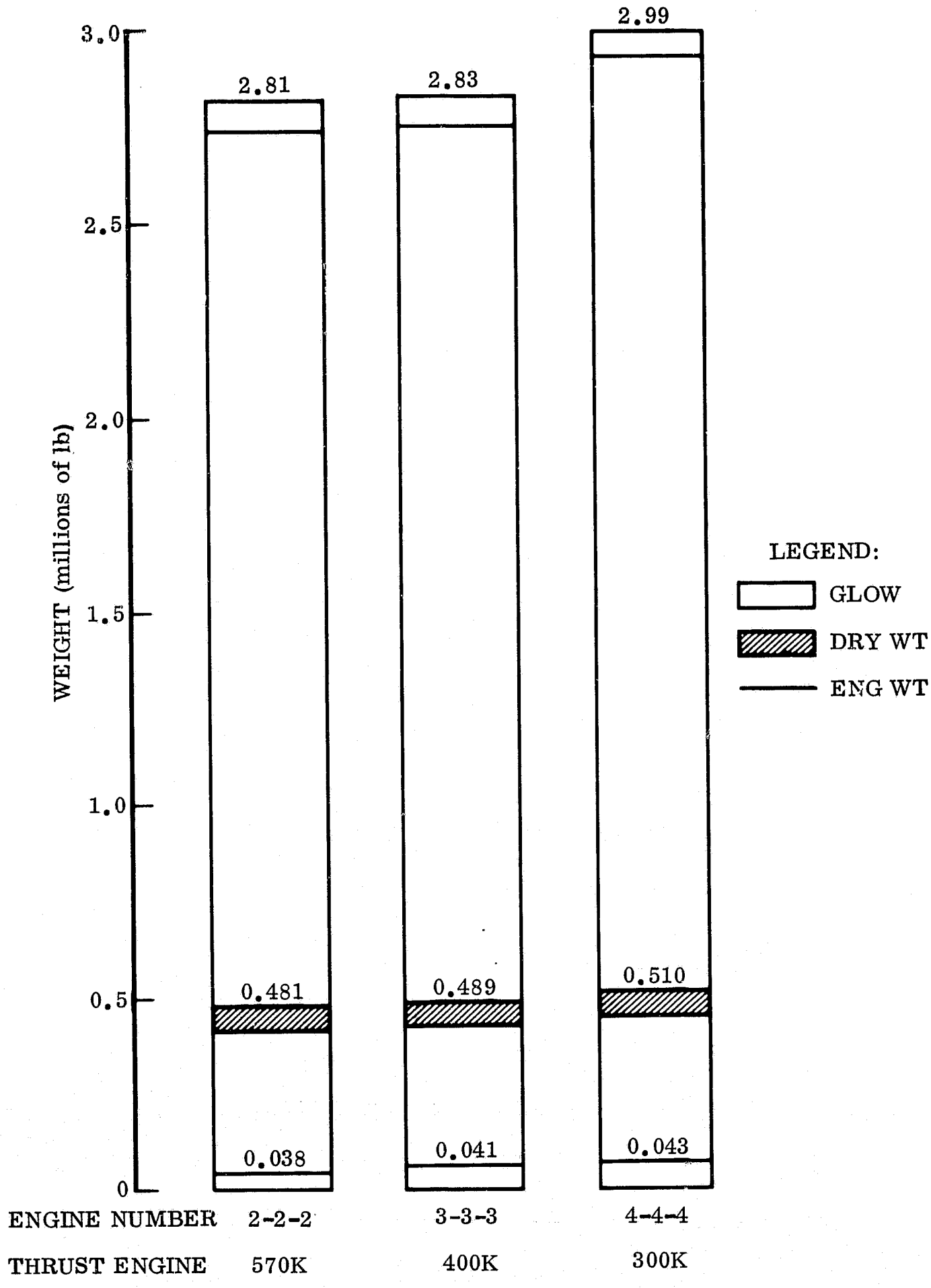


Figure 2-44. Summarized Weights for the 25K-Lb Payload FR-1 System

2.2.6.1 Test Programs. The ground test program would be more affected than the flight test program by a change in engine size and number. For instance, component and subsystem development laboratories would have a slight increase in the number of individual hardware components for evaluation, qualification, and validation testing with the lower thrust engines. Thrust vector control (TVC) subsystem tests would be more complex and would require increased test time. Tail section structural loads tests may also be extended to thoroughly explore all combinations of loads from the various thrust load points under various engine-out conditions. Propulsion/vehicle integration testing would be affected by a difference in the number of subsystem components and require additional test time to adequately evaluate all likely operational conditions and failure conditions.

Even though these major vehicle ground tests are affected by engine selection, their overall effect is considered insignificant in view of the total program. Slight variations in the applicable ground test facilities would be required to accommodate the particular engine selected, but assuming this selection preceded facility construction, there would be no relative effect to consider.

The flight test program is relatively unaffected except for the propulsion system prelaunch checkout times.

For both the ground and flight test programs, the effect of engine selection, although insignificant overall, would be noticed in component quantities, some extended tests, instrumentation requirements, and data evaluation.

2.2.6.2 Manufacturing. Some specific detail and assembly tools would be affected by engine selection and total assembly times and sequences may alter slightly; these are again judged to be insignificant for the test article and production vehicle rates appropriate for the space shuttle program. Any effect on other manufacturing facilities are likely nonexistent.

2.2.6.3 Operations and Support. Effect of engine selection on operational facilities and handling equipment is considered negligible and would likely be limited to accessibility to the vehicle base hard points for handling and launch erection.

A more noticeable effect would be associated with postflight turnaround time. The lower-thrust version requiring more engines would cause an increase in postflight inspection activity relative to the propulsion system. This would be reflected by increased manpower or inspection and maintenance time, or both, due to the additional quantity of hardware components and engines. Inspection time is affected by both hardware component size and quantity while overhaul or remove-and-replace maintenance activities are more sensitive to hardware quantity. As a result, postflight turnaround time and cost will experience some increase, but likely not enough to affect total vehicle inventories.

2.2.6.4 FR-1/Two-Element Comparison. The foregoing statements are generally applicable to either the FR-1 or the two-element space shuttles when considering an increase or decrease in the number of engines involved, and are more noticeable with the booster elements than for the orbiters. The particular program effects, as noted above for each engine selection, would always tend to be more pronounced for the two-element booster than for the three-element booster due solely to the number of engines and quantities of support hardware involved.

2.2.7 RELIABILITY AND SAFETY. A reliability study was conducted to aid in the selection of engine size, using the two-element sequential burn FR-3 vehicle and the parallel burn FR-1 as models. Engine reliabilities of 0.99 and 0.997 were considered. The probability of loss of engine(s) was determined for the several engine clusters evolved. Probability of loss of thrust and catastrophic failure of the engines (explosions) were separately considered.

2.2.7.1 Groundrules and Assumptions. The following groundrules and assumptions were applied uniformly for all cases analyzed:

a. Burn-in type failures are to be eliminated through adequate operation and check-out of engines and associated propellant feed hardware.

b. Wear-out failures should be minimized by using components with adequate design life, substantiated by testing. State-of-the-art components that are wear-out limited will require special consideration for time-based replacement and spares provisioning.

c. A constant failure rate is assumed for the comparative reliability analysis.

d. Engine operating time during countdown and boost was 0.06 hour.

e. MTBFs of 15 and 50 were used in the analysis. With total operating time approximately 0.155 hours, these yield engine reliabilities of 0.99 and 0.997 respectively.

f. Minimum thrust/weight ratios of 1.16 and 0.825 for booster and orbiter, respectively, are required for abort safety with one engine out.

2.2.7.2 Results. Table 2-18 is a summary of probability of engine losses for FR-3 and FR-1 vehicle systems for an engine reliability of 0.99. For the 300K lb sea level thrust engine, minimum liftoff F/W of 1.16 can be achieved in both vehicle systems with up to three engines out. The 400K lb engined FR-1 system also achieves F/W 1.16 with three engines out at liftoff. The 400K lb engined two-element sequential vehicle system and the 500K-lb FR-1 have reduced engine-out capability at liftoff, the minimum F/W of 1.16 resulting from two engines out. All other vehicle designs (incorporating 500K, 600K, 1000K lb sea level thrust engines) meet the F/W = 1.16 rule with one engine out.

Table 2-18. Summary — Probability of Loss of Engines/1000 Flights ($R_{ENG} = 0.99$)

Engine Size	FR-3			FR-1		
	1 Engine Out	2-3 Engines Out Noncatastrophic	Catastrophic Failures	1 Engine Out	2-3 Engine Out Noncatastrophic	Catastrophic Failures
300K	85	< 0.01 (Orbiter^{oo} 3 Out) < 0.01 (Booster***3 Out)	0.80	104	< 0.01 (Orbiter^{00} 3 Out) < 0.01 (Boosters***3 Out)	1.00
400K	63	0.05 (Orbiter^{o} 2 Out) 0.1 (Booster** 3 Out)	0.60	79	< 0.01 (Orbiter^{oo}3 Out) < 0.01 (Booster***3 Out)	0.80
500K	50	0.05 (Oribter^{o} 2 Out) 0.7 (Booster* 2 Out)	0.50	60	< 0.01 (Orbiter^{oo}3 Out) 0.055 (Booster** 3 Out)	0.60
602K	40	0.016 (Orbiter^{o} 2 Out) 0.50 (Booster* 2 Out)	0.40	—	—	—
984K	—	—	—	32	0.016 (Orbiter^{o} 2 Out) 0.25 (Booster* 2 Out)	0.30

***L.O. T/W >1.16 with three engines out
**L.O. T/W >1.16 with two engines out
*L.O. T/W >1.16 with one engine out

^{oo}Staging T/W > 0.825 with two engines out
^{o}Staging T/W > 0.825 with one engine out

Catastrophic failure data given in Table 2-18 was obtained from an analysis summarized in Table 2-19. This data was generated using an assumed catastrophic failure probability of one percent of the basic engine failure rate. Failures/1000 flights are shown for the two engine reliabilities for FR-3 and the FR-1. As shown, failures are linear with number of engines, with the least number occurring with higher thrust engines.

2.2.7.3 One Engine Out (Representing Mission Losses). Larger engine modules (lesser numbers of engines) reduce the probability of single-engine failure, which reduces mission losses from 85 (300K module) to 32 (984K module). Mission losses are a minor operational cost factor and reasonable mission losses from all causes $\approx 50/1000$ flights. Engine failure contribution to this should be ≈ 10. A module size above 500K results in least mission losses.

2.2.7.4 Two/Three Engines Out (Representing Vehicle Losses and Safety). Larger engine modules (lesser numbers of engines) reduce the probability of double or triple engine failure. These engine failures can result in vehicle losses, depending on liftoff and staging F/W in a particular vehicle configuration. Vehicle losses are a major operational safety and cost consideration. Less than 0.1 vehicle losses per 1000 flights represents an adequate safety goal and the engine failure contribution to this should be

Table 2-19. Catastrophic Failures of Engines — Assumed Failures per 1,000 Flights (Ascent Phase) = 1% of Engine Failures

FR-3			FR-1		
	Failures/1000 Flights			Failures/1000 Flights	
Engines	$R_{ENG} = 0.99$	$R_{ENG} = 0.997$	Engines	$R_{ENG} = 0.99$	$R_{ENG} = 0.997$
300K (17/4)	0.8 (1,250)*	0.25 (4,000)	300K (7/6/7)	1.0 (1,000)	0.31 (3,720)
400K (13/3)	0.6 (1,660)	0.2 (5,000)	400K (5/5/5)	0.8 (1,250)	0.24 (4,160)
500K (10/3)	0.5 (2,000)	0.16 (6,250)	500K (4/4/4)	0.6 (1,660)	0.2 (5,000)
602K (8/2)	0.4 (2,500)	0.12 (8,326)	984K (2/2/2)	0.3 (3,333)	0.009 (11,100)

*(Flights/Failures)

less than <0.03/1000. In the two-element configuration, vehicle losses are approximately the same when considering noncatastrophic and catastrophic engine failure. However, a slight adjustment in F/W at liftoff to provide the minimum value with two engines out for the 500K and 602K engines would result in substantially lower vehicle losses from noncatastrophic type failures.

The FR-1 vehicles with the 300K modules have a three-engine out liftoff capability, but losses are greater than the vehicle (with 900K) limited to a one-engine out capability because increasing numbers of engines increase catastrophic failures linearly, and catastrophic failures represent vehicle losses. With the F/W adjustment in the two-element vehicle, and considering both vehicles, module sizes between 500K and 900K tend to show lowest vehicle losses.

2.2.8 COST ANALYSIS

2.2.8.1 Approach. A cost analysis was conducted using the 50,000-lb payload FR-1 and FR-3 vehicles as baseline configurations. The baseline FR-1 vehicle had six 984,000-lb thrust engines; the FR-3 had 8 booster and 2 orbiter engines at 602,000-lb thrust. This analysis consisted of determining the differences in total program costs resulting from substituting 300K, 400K and 500K-lb thrust engines in the baseline vehicles.

Groundrules and assumptions used in developing vehicle cost data are shown in Tables 2-20, 2-21, and 2-22. It should be noted that this cost analysis was conducted on point designs synthesized for 300K, 400K, and 500K-lb thrust engines rather than on a parametric study using rubberized engine sizes. Because of this, the resulting cost data exhibits some step functions instead of smooth trends.

2.2.8.2 Vehicle Cost Comparisons. Total program costs generated for the FR-1 and FR-3 baseline vehicles are shown in Figure 2-45. These costs are based on 50 flights per year for 10 years of operations.

Costs associated with the use of various rocket engine thrust levels and numbers of engines were determined using the same methodology as used for the baseline vehicle costs. For each candidate vehicle configuration it was assumed that a single rocket engine configuration would be developed and used for both the orbiter and booster stages. Results of this exercise are shown in Figures 2-46 and 2-47. The data is presented as changes in cost from the baseline vehicle program costs shown in Figure 2-45. Table 2-22 shows the changes in cost from the baseline vehicle for the FR-3 vehicle configurations. There appears to be no general total program cost trend as a function of rocket engine thrust level. Program cost for the 500K-lb thrust engine configuration shows an increase in development cost as well as investment and

Table 2-20. Development Cost Basis

		FR-1		FR-3	
		Orbiter Elements	Booster Elements	Orbiter Elements	Booster Elements
1.	Laboratory Hardware Consumption (Basic Body Systems Only)	2		2	2
2.	Ground Test Program Personnel Requirements (Man-Months)	6,300		9,500	
3.	Major Ground Test Hardware Consumption (Complete Elements, Exclusive of Propulsion)	6.2		5.2	4.1
4.	Number of Equivalent Full-Duration Flight Tests	14	15	16	21
5.	Flight Test Program Personnel Requirements (Man-Months)	5,500		6,500	
6.	Flight Test Hardware Consumption* (Complete Elements Including Propulsion)	2	1	2	1
7.	Flight Test Spares & Refurbishment Materials Consumption (Complete Elements Including Propulsion)	2.8	2.8	2.8	1.8
8.	Number of Launch Pads Constructed	2		2	
9.	AGE Requirements (No. Ship Sets)	4		4	
10.	AGE Spares Requirement (% of Total Ship Sets)	16%		16%	

*Excludes one FR-1 orbiter, two FR-1 boosters, two FR-3 orbiters and two FR-3 booster elements transferred to operating inventory.

operations cost. This can be explained in the following manner. Mission performance requirements could be satisfied with a 10-engine booster and 2-engine orbiter configuration, but because of the one-engine-out, once-around mission abort requirement, an extra engine was required on the orbiter. This increased the vehicle unit hardware cost and, therefore, the cost of development hardware. A slightly higher-thrust engine could have met all design requirements at a lower total program cost.

As can be seen in Figure 2-47, (FR-1 configuration) there appears to be a general trend toward lower total program cost as larger numbers of lower-thrust engines are used. A more detailed study of engine-associated operations and maintenance costs and an analysis of the effects of reliability on cost for multiple engine configurations may, however, reduce or eliminate this trend.

Table 2-21. Investment Cost Basis

		FR-1		FR-3	
		Orbiter Elements	Booster Elements	Orbiter Elements	Booster Elements
1.	Number of Inventory Elements Procured* 50 Launches/Year	4	6	4	4
2.	Initial Spares (% of Inventory Hardware Cost)	10%	10%	10%	10%
3.	Number of Operational Ship Sets of AGE Procured**	1		1	
4.	Number of Additional Launch Pad Facilities Contructed	1		1	

*Includes one FR-1 orbiter, two FR-1 boosters, two FR-3 orbiters and two FR-3 booster elements transferred from development program.

**Two additional complete sets of AGE were assumed to be available at no cost from development program.

Table 2-22. Operating Costs Basis

	Cost Items	Costing Basis
1.	Personnel Costs (Excluding Rocket Engine Contractor Personnel):	
	Operating Personnel	$30,000/man year
	Support Services	$15,000/man year
2.	Rocket Engine Overhaul & Maintenance (Materials & Personnel Costs)	25% of engine cost for every 100 engine flights
3.	Jet Engine Overhaul & Maintenance Materials	0.5% of engine cost/flight
4.	Thermal Protection System Maintenance & Repair Materials	0.313% of TPS unit cost/flight (booster) 2.140% of TPS unit cost/flight (orbiter)
5.	Avionics System Maintenance & Repair Materials	0.5% of avionics cost/flight
6.	Airframe Structure Maintenance & Repair Materials	0.035% of structure unit cost/flight
7.	Airframe, Subsystems Maintenance & Repair Materials	0.5% of subsystems cost/flight

Table 2-22. Operating Costs Basis, Contd

Cost Items	Costing Basis
8. Propellants & Gases	\$0.15/lb of LO_2/LH_2 mix applied to total propellant weight
9. GSE Maintenance	5% of operational GSE total procurement cost/year
10. Facilities Maintenance	5% of operation facilities total procurement cost/year

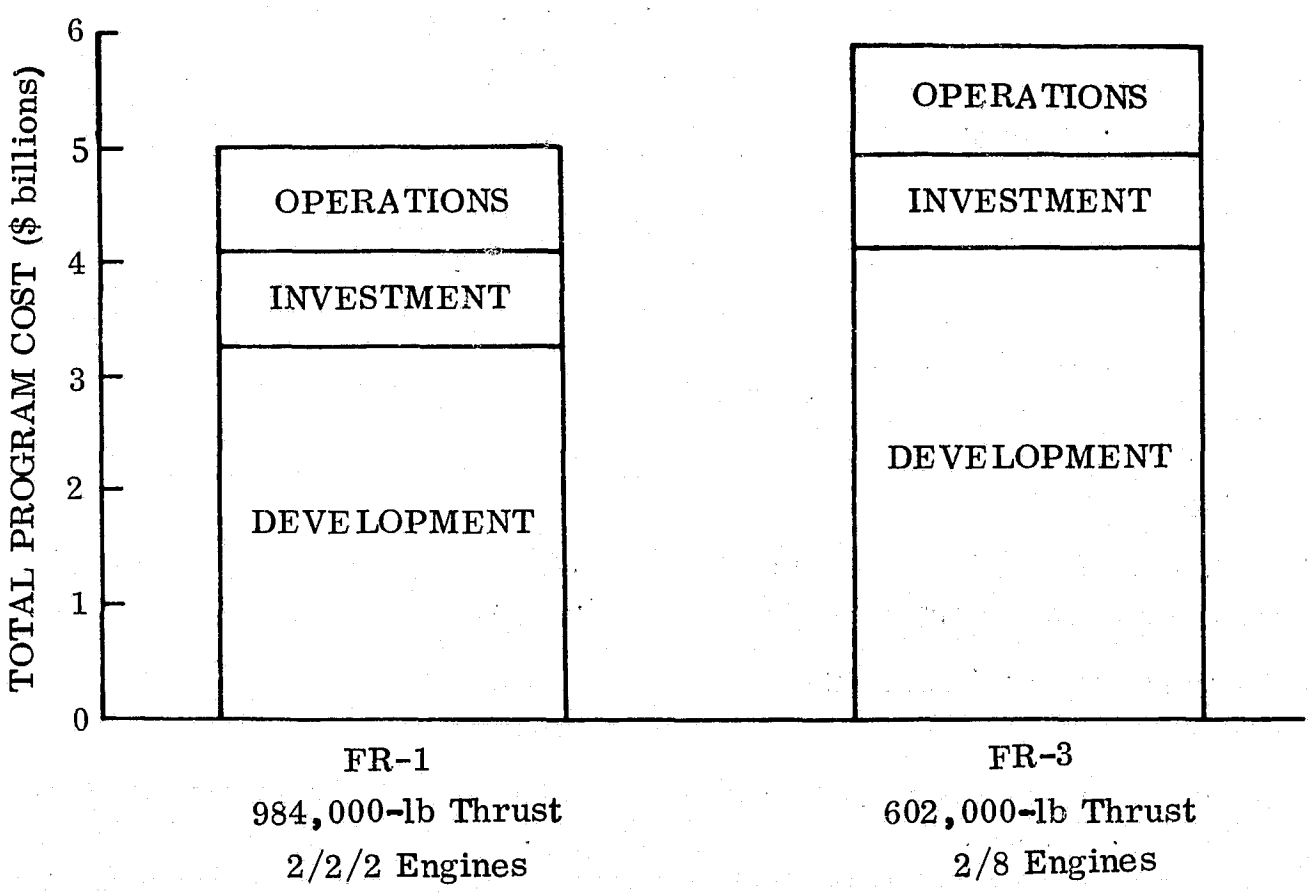
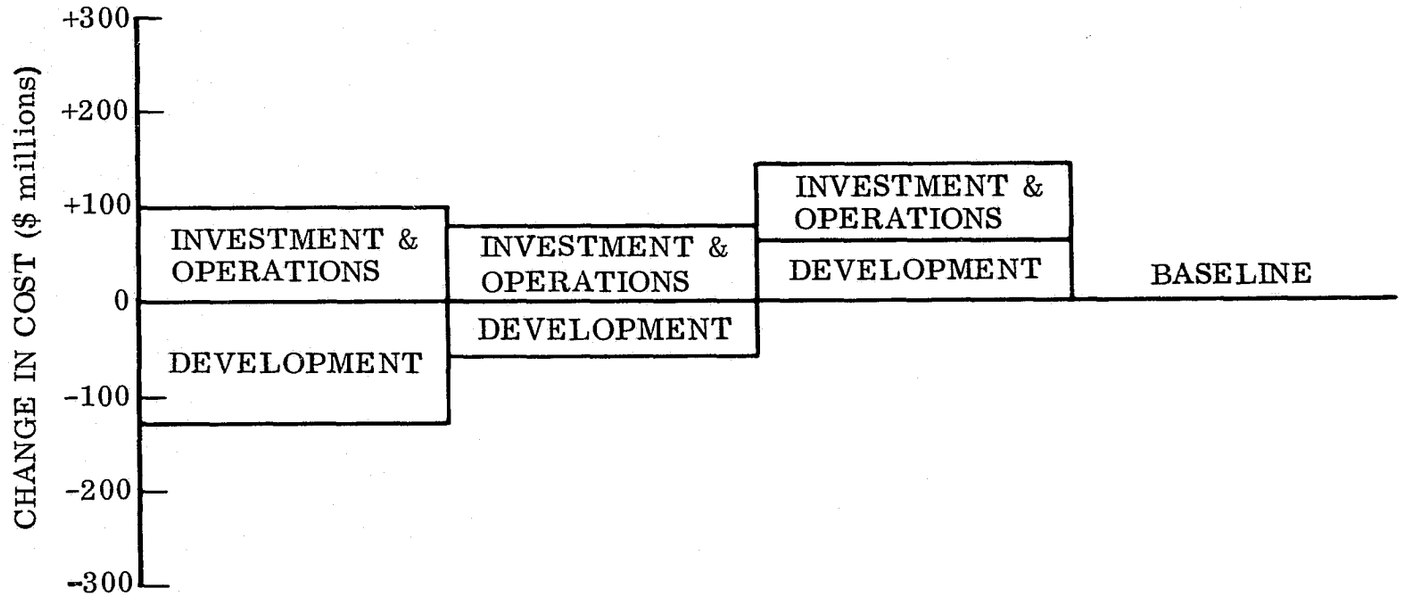
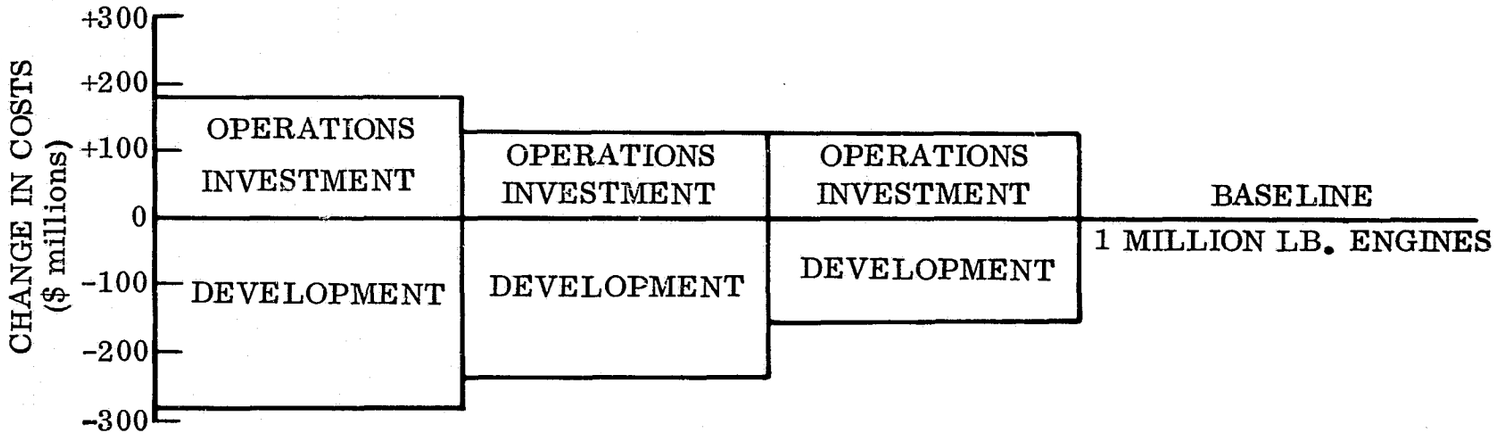


Figure 2-45. Total Program Cost of Baseline Vehicle Configurations (50,000 Lb Payload; 50 Launches Per Year)



SEA LEVEL THRUST/ENG. (LB)	300,000	400,000	500,000	602,000
NO. ENGINES/BOOSTER	17	13	10	8
NO. ENGINES/ORBITER	4	3	3	2
NET ΔCOSTS ($ M)	-24	+24	+147	—

Figure 2-46. Total Program Cost versus Rocket Engine Thrust Level + Number of Engines — Fifty Launches per Year — 50,000 Lb Payload FR-3 Vehicle



SEA LEVEL THRUST/ENGINE (LB)	300,000	400,000	500,000	984,000
NO. ENGINES/BOOSTER	7	5	4	2
NO. ENGINES/ORBITER	6	5	4	2
NET ΔCOSTS ($ M)	-111	-110	-38	—

Figure 2-47. Total Program Cost versus Rocket Engine Thrust Level + Number of Engines — Fifty Launches per Year — 50,000 Lb Payload FR-1 Vehicle

Program costs generated for 100 launches/year indicate that the cost advantage of using a large number of low-thrust engines is associated with low launch rates. Higher launch rates tend to favor the use of fewer high-thrust engines.

2.2.9 EVALUATION. The data generated on cost, weight, performance, and failure rates, together with overall vehicle layouts, was examined to establish which engine-thrust class was preferable.

It is apparent that the weights and performance figures show very little change across the engine thrust value spectrum. The same is true for total program costs. The FR-1 system does show about a two percent reduction for the system with the 7-6-7 arrangement of 300K engines. However, it is felt that the increased complexity in this system would tend to increase weights and costs above the present iteration results and that essentially this two percent saving must be considered to be overshadowed by the tolerances of the analysis. Neither weights nor costs, then, give a basis for selection.

Examination of failure rate data given in Section 2.2.7 indicates that large numbers of smaller thrust engines will incur higher probability of failure than smaller numbers of higher thrust engines. This will lead to greater numbers of aborted missions, since there is no redundancy in engines, and with one engine out the mission cannot be completed. The vehicle will be returned intact, however, since all have intact abort capability with one engine out. The only penalty for one engine out will be a return to base and an approximate two-million-dollar turnaround cost in preparation for the next mission. Some of the vehicles with larger numbers of engines have capability to fail safe with two or even three engines out. While on the face of it this appears to add up to greater vehicle safety, this is not really true since catastrophic engine failures determine vehicle safety. The assumption that one percent of all engine failures lead to explosions and loss of the vehicle reverses the pattern. It would still be true, however, if the catastrophic failure rate was much less than one percent. For example, consider the case of the 400K-lb thrust, 13/3 engine, 50K-lb payload, two-element sequential burn vehicle. At 0.99 engine reliability, with either two booster engines or one orbiter engine out, a safe mission abort and intact return to base is possible. Engine failure rate data indicates a mission abort rate of 63 per 1000 and a vehicle loss rate of 0.1 per 1000 without catastrophic failure. However, the catastrophic engine failure assumption of one percent says the vehicle loss rate is 0.63 per 1000 or six times greater than indicated by the engine-out probability alone. In the light of catastrophic loss trends, a small number of higher thrust engines is indicated.

With regard to complexity, more engines require more connections, more intersections, more complicated line routing and higher non-optimum factors for flange points, transitions, penetration-of-thrust-structure webs and firewalls, and generally an increase in weight as the design is evolved in detail. The complexity increases in the control subsystem (two actuators and servos per engine), in propellant feed (two PVC ducts per engine), and in overall maintenance and accessibility of aft-end components.

All design experience and intuition tends to lead in the direction of as few engines as possible consistent with engine-out safety and the maximum size engine that can conveniently be developed. This is generally the historical precedent in aircraft and existing boosters.

2.2.10 CONCLUSIONS

2.2.10.1 Vehicle/Engine Design Integration. Installation of 300K, 400K, and 500K-lb rocket engine modules in the FR-1 and FR-3 vehicle systems appears to be feasible. As the numbers of engines increase, relative complexity also increases.

Relations between fuel tank aft bulkhead geometry, thrust structure, engine requirements and propellant feed ducting are the major items that determine volumetric requirements and relative complexities of the aft section of the vehicles. Since the vehicles are reusable, special consideration must be given to accessibility. For those cases involving engine quantities up to approximately six, accessibility is usually satisfied by the standard clearances and space requirements of the major component envelopes. As the number of engines increase, basic clearances fail to provide adequate accessibility; therefore, the aft section of the vehicle must be increased. The degree of feed ducting complexity is related to available space. For vehicles involving three or four engines, manifolds can be simple without penalties to adjacent structure or to the flow characteristics. Increase in the number of engines, however, reduces the space envelopes which in turn cause the basic manifold routings to become sensitive to adjacent structure and equipment. Ducting configurations are generally more complex when the number of feed points is increased, and this degree of complexity is augmented when simple routings have to be diverted for clearances. For sequential-type vehicles, the absence of cross-flow requirements simplifies propellant feed manifolding.

2.2.10.2 Weight and Cost Comparison. Installation of 300K, 400K, and 500K-lb rocket engine modules in the 50K-lb payload FR-1 and FR-3 vehicle systems does not involve any marked change in system weights or costs relative to the larger-thrust baseline cases.

2.2.10.3 Vehicle Afterbody Effects. Convair wind tunnel tests have shown that a blunt-base vehicle does not involve severe penalties to the vehicle system in terms of booster trimmed flyback L/D. With a blunt-base vehicle, a two-engine configuration does not allow best usage of the available base area in terms of rocket engine expansion ratio and specific impulse. Allowing for gimbal clearance it appears that a four- or a three-engine arrangement will result in higher expansion ratios and higher overall specific impulse. Above four engines the clearance requirements tend to restrict the expansion ratio again. Installation of lower thrust, shorter length engine modules allows easier heat-shielding of the engines at entry by the elevons and fairings across the bottom of the vehicle base. For these reasons a three- or four-engine

element is a very feasible arrangement. This applies to the FR-1 booster or orbiter or the FR-3 orbiter element. The two-stage booster of course requires many more than four engines.

2.2.10.4 Engine Thrust Levels. The number of engines should be kept low in order to maximize mission successes and to minimize catastrophic engine failures and vehicle losses. For this reason the 300K and 400K-lb thrust level engines are not recommended since no cost or performance advantage is indicated from their use.

With respect to the orbiter once-around abort analysis, current indications are that the 2-2-2 FR-1 (50K) is marginal. The 4-4-4 FR-1 will have a comfortable reserve in this respect. Therefore, the 4-4-4 FR-1 with 500K-lb sea level nominal thrust engines is recommended. (A 3-3-3, 50K-lb payload FR-1 would also be attractive, although this is outside the thrust range of this study and was therefore not examined.)

The sequential burn two-element system is less affected by the orbiter abort criteria; however, the 10-3 arrangement of 500K-lb thrust engines is selected over the baseline, which is an 8-2 arrangement of 600K-lb sea level nominal thrust engines, in order to keep the engine size down and to give a greater abort margin in the orbiter with one engine out.

In the case of the 25K-lb payload FR-1 system with a baseline thrust of 570K-lb sea level thrust engines in a 2-2-2 configuration, there is no reasonable installation for the 500K-lb module engines. A 3-3-3 installation of 400K-lb engines is recommended for the 25K-lb payload FR-1. This appears to be a reasonable candidate; however, at NASA request, no costs were run on the multi-engine 25K-lb payload systems.

2.3 COMPARISON OF BELL AND AEROSPIKE ENGINES

During the course of the ILRV study, a number of comparisons were made between the high P_c bell nozzle and the aerospike engine as new NASA direction was received or the aerospike engine definition was modified. These comparisons are briefly described below:

a. Optimized bell and integrated round dual-combustor aerospike engine installations in FR-1 vehicles sized for the 50,000-lb payload were compared. Chamber pressures were 2,000 psi for aerospike and 3,000 psi for bell engines. Aerospike engines studied were single-combustor and dual-combustor round configurations, and a dual-combustor oval configuration tailored to fit a boattailed vehicle base.

b. Use of 2,000 psi button aerospike and 3,000 psi bell engines in fixed-weight FR-1 vehicles was compared. A button aerospike was used on a vehicle optimized for use of the high P_c bell engine and not compromised in any way for the ultimate use of the aerospike. Payload was allowed to vary with the aerospike engine.

c. The use of 2500 psi aerospike and 3000 psi bell engines was compared in FR-4 non-crossfeed vehicles with launch weight fixed at 3,000,000 lb. Optimized bell, optimized integrated aerospike, and optimized button aerospike designs were developed.

The integrated aerospike is configured to almost completely fill the vehicle base area, giving maximum possible area ratio for a given thrust and chamber pressure. In the dual-combustor version, total engine flow is divided between two closely nested concentric annular combustors. The outer combustor can be throttled, increasing the effective area ratio and vacuum I_{sp}. For an equal flow split between combustors, area ratio is almost doubled and I_{sp} is increased approximately 10 seconds when the outer combustor is completely cut off. Typical area ratio and I_{sp} values for dual-chamber aerospike and high P_c bell engines with two-position nozzles are as follows:

	Aerospike	Bell
Sea Level I_{sp} (Low ϵ), sec	370.5	384
Vacuum I_{sp} (Low ϵ), sec	449.	-
Vacuum I_{sp} (High ϵ), sec	460.0	450.5
Low Area Ratio	103	35
High Area Ratio	198	102

The throttling profile (percent thrust versus time) for a typical mission is such that the higher I_{sp} values for the aerospike can be obtained for only limited times near the end of booster and orbiter solo phases of flight. For the bell, the high value of I_{sp} is effectively reached by 100 seconds into the flight, and is maintained to injection.

The button aerospike is designed so that it can operate within the same envelope as the high P_c bell engine, and interfaces (gimbal block, gimbal actuator attach points, etc.) with the vehicle are similar. The button aerospike does not utilize base area as fully as does the integrated engine, so its area ratios and vacuum I_{sp} are somewhat lower. Button aerospike engines can also be made with either single or dual combustors.

Results of these studies are summarized in Table 2-23. Performance of the 2000 psi dual-chamber integrated aerospike and high P_c bell engines is very nearly the same in FR-1 vehicles using a common main propulsion system in all elements. As can be seen, vehicles sized for the 50,000-lb payload, using either propulsion system, weigh approximately the same. The single-chamber aerospike gave somewhat lower performance, increasing launch weight 200,000 lb (3.8 percent compared to the bell).

Performance for vehicles using the 2000 psi button aerospike engine is inferior to that for the high P_c bell engine. Replacement of the bell with button aerospike engines in a vehicle designed for 50,000-lb payload using the bell engine reduced payload to less than 33,000-lb.

Performance using the 2500 psi integrated aerospike (dual-chamber in boosters, single-chamber in orbiter) is superior to that using bell engines. A vehicle constrained to a launch weight of 3,000,000-lb and using the integrated aerospike had almost double the payload of a vehicle using the bell, if penalties for use of secondary injection were not considered. Magnitude of these penalties has not been assessed.

Performance for vehicles using the 2500 psi button aerospike is nearly as high as for vehicles using the bell. A vehicle constrained to a launch weight of 3,000,000-lb had approximately 85 percent of the payload for a similar vehicle using the bell engine.

Table 2-23. Summary of Bell and Aerospike Comparisons

Comparison	Engine Type	Payload (lb)	Launch Weight (lb)	Comments
Bell vs Integrated Aerospike (2000 psi) in FR-1 Parallel Burn Vehicle with Fixed Payload	Bell (2-2-2)	50,000	5,330,087	1. Aerospike engines use secondary injection for thrust vector control. Vehicles not penalized for extra propellant required for injection.
	Dual-Combustor Aerospike, Round	50,000	5,356,621	
	Dual-Combustor Aerospike, Oval	50,000	5,459,431	2. Oval aerospike tailored to fit boattail of FR-1 vehicle.
	Single-Combustor Aerospike, Round	50,000	5,529,311	
Bell vs Button Aerospike (2000 psi) in FR-1 Parallel Burn Vehicle with Fixed Launch Weight	Bell (2-2-2)	50,000	4,535,165	1. Bell and button aerospike engine gimballed for TVC.
	Dual-Combustor Button Aerospike (2-2-2)	<33,000	4,502,165	
Bell vs Integrated and Button Aerospike (2500 psi) in FR-4 Sequential Burn Vehicle with Fixed Launch Weight	Bell (6-3-6)	15,000	3,000,000	1. Bell and button aerospike engine gimballed for TVC.
	Button Aerospike (3-3-3) Dual-Combustor in Booster	12,500	3,000,000	
	Integrated Aerospike Dual Combustor in Booster	28,000	3,000,000	2. Secondary injection for thrust vector control. Vehicles not penalized for extra propellant required for injection.

2.4 PROPULSION DESIGN FOR FINAL FR-3 TWO-STAGE SEQUENTIAL BURN VEHICLE

The booster and orbiter elements of the FR-3 have three propulsion subsystems, i.e., main propulsion, the attitude control propulsion subsystem (ACPS), and airbreathing (flyback) propulsion, which are described in Sections 2.4.1, 2.4.2, and 2.4.3, respectively.

2.4.1 MAIN PROPULSION. Main propulsion for the FR-3 is provided by high P_c hydrogen-oxygen bell-nozzle engines. The main engines operate in a pump-fed mode during boost and on orbit to provide orbit maneuvers.

Propellants are supplied to the engines by insulated propellant lines from internally insulated tanks. The tanks are pressurized with warm gas to minimize residuals, using gaseous hydrogen for the liquid hydrogen tanks for the booster and orbiter. High pressure helium is used to pressurize the oxygen tank in the booster, and gaseous oxygen is used to pressurize the orbiter liquid oxygen tank.

Booster engine area ratios are 35/80, operating retracted at an area ratio of 35 to an altitude of 30,000 feet, where all nozzles are extended to take advantage of the higher area ratio. The orbiter engine has an area ratio of 35/160, the higher area ratio being utilized to provide the higher I_{sp} desired by the upperstage. Very recent studies indicate that expansion ratios greater than 160 (e.g., 200) will result in increased payload.

The base nozzle contour defined by Pratt and Whitney is used. Using this contour instead of the maximum performance contour increases launch weight less than 0.3 percent, yet decreases engine length by three feet. The shorter nozzle is easier to protect from aerodynamic loads during ascent and from heat during entry. Engine performance for the FR-3 vehicle is summarized in Table 2-24.

The nominal engine mixture ratio is 6.5. This mixture ratio selection was based on a tradeoff considering the lower structural weight of higher mixture ratios and the higher specific impulse and therefore lower propellant weight at lower mixture ratios. During ascent, the engine mixture may be controlled to minimize residual (i.e., propellant utilization). A mixture ratio range of ± 0.35 around the nominal of 6.5 is sufficient for control. This results in operation of the engine well within the engine MR operating constraints of 6 and 7.

During the initial phase of ascent, the fifteen booster engines operate at maximum nominal thrust providing a liftoff F/W of 1.387. Should an engine become inoperative (outside specification limits) the other engine will be operated at 108 percent of maximum nominal thrust. This rating provides maximum payload consistent with propellant utilization control range requirements. When the vehicle reaches a

Table 2-24. FR-3 Engine Performance

	Booster	Orbiter
Number of Engines	15	3
Sea Level Thrust (lb)	400,000	-
Vacuum Thrust (lb)	462,000	472,000
Specific Impulse, Sea Level (sec)	See Confidential Addendum	
Vacuum (sec)		
Area Ratio, Sea Level		
Vacuum		
Mixture Ratio		
LH_2-NPSH 100%F	60	60
20%F	0	0
LO_2-NPSH 100%F	16	16
20%F	5	5
Engine Weight	4400	4600
Type Nozzle	Base	Base

maximum acceleration of 3g the engines are throttled to maintain 3g. Engine throttling is initiated at approximately 100 seconds after liftoff. Approximately 5 seconds prior to shutdown, the engines are throttled from approximately 60 percent thrust down to 10 percent thrust to minimize residuals. The booster stages at approximately 190 seconds after liftoff.

A few seconds prior to booster engine shutdown the three orbiter engines are started. The orbiter engine start sequence results in the orbiter vehicle achieving full thrust nominally at stage separation. The orbiter thrust to weight ratio after stage separation is 1.52. The orbiter accelerates until it reaches 3 g at approximately 330 seconds. The orbiter begins to throttle during the remainder of the orbiter solo phase. The orbiter is throttled to 70% by 390 seconds, just prior to propellant depletion. Within the last five seconds prior to shutdown the engine is further throttled to 10% to achieve minimum residuals.

After shutdown, the orbiter coasts from 40 n. mi. to 100 n. mi. During this time liquid residuals tend to keep the pump cool. When the orbiter reaches 100 n. mi. the first orbit maneuver is accomplished to circularize the orbit.

For orbit maneuvers, one orbiter engine is used in a pump-fed mode at 10 percent of the maximum throttle setting, i.e., 47,000-lb thrust. Propellants for the orbit maneuvers are provided to the engine from maneuver tanks through lines that are insulated to minimize boiloff.

Before the first orbit-maneuver firing, the oxygen-hydrogen ACPS thrusters in the rear of the vehicle are fired for approximately two minutes, providing a thrust of 10,000 pounds to settle the propellant, i.e., to provide liquid to the engine pumps. At the initiation of settling the prevalve of the secondary (maneuver) propellant subsystem is opened, allowing the propellants to flow to the main engine through the secondary lines.

During settling for the circularization maneuver at 100 n.mi., a ΔV of approximately 75 fps is provided by the ACPS. The main engine therefore provides only 37 fps for the first maneuver. The main engine burns approximately 10 seconds at 10% thrust to provide this ΔV.

After the first orbit maneuver, gaseous propellant is fed to a port downstream of the pump discharge. The gas forces liquid in the pump and secondary propellant line back into the tank. After all the liquid is pushed into the maneuver tank a prevalve is closed, maintaining the liquid in the highly insulated tanks.

After circularization, the orbiter may be required to remain in the 100 n.mi. orbit for up to 18 hours to permit phasing. During this time the engines and their pumps will receive heat from the sun, earth, and vehicle, causing the pump temperatures to increase.

After the coast phase a second orbit maneuver is required to transfer the vehicle from 100 n.mi. to 260 n.mi. The second maneuver starts by operating the ACPS to settle the propellant in the maneuver propellant lines. ACPS settling is approximately three minutes prior to main engine start.

Two minutes prior to main engine start, chilldown flow through the main engine pumps cools the pumps to near liquid-propellant temperature. Approximately 420 pounds of oxygen and 170 pounds of hydrogen are used to chill down the pumps. Main engines are started during the last five seconds of the attitude control settling phase. The main engine maneuver is accomplished in approximately 100 seconds. After main engine shutdown, the propellant is again pressurized out of the secondary line back into the maneuver tanks.

The third and fourth maneuvers, i.e., the maneuver to circularize at 260 n.mi. and the maneuver to deorbit the vehicle, are accomplished in a manner similar to the second maneuver. Main engine operating times for these maneuvers are 110 and 130 seconds, respectively.

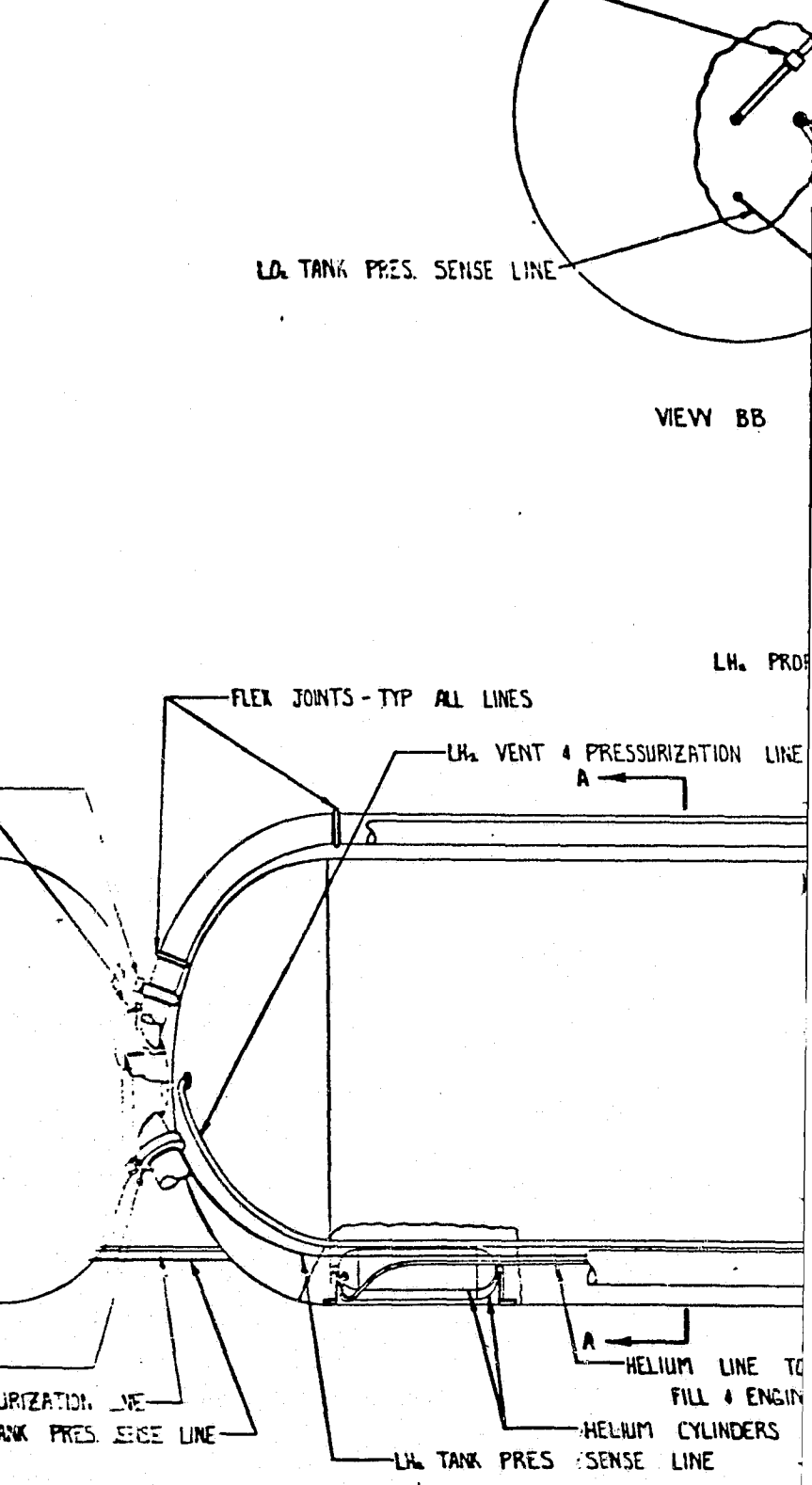
2.4.1.1 Booster Propellant Feed. The propellant feed subsystem consists of all propellant ducting, pressurization subsystems, and venting subsystems required to condition propellants and supply them to the booster and orbiter engines, both for boost and orbit-maneuver firings. Figure 2-48 is a detail layout of the FR-3 booster design.

2.4.1.1.1 Liquid Oxygen Subsystem. The LO_2 tanks are located forward so that the vehicle is aerodynamically stable during ascent. Long main LO_2 lines are required to convey LO_2 to the engine manifold ducting. LO_2 is withdrawn from a common central sump under a flat-plate baffle which prevents vortexing and minimizes residuals at depletion. Multiple main LO_2 lines are used to reduce the magnitude of vehicle/engine development (integrated testing) and to simplify manifolding.

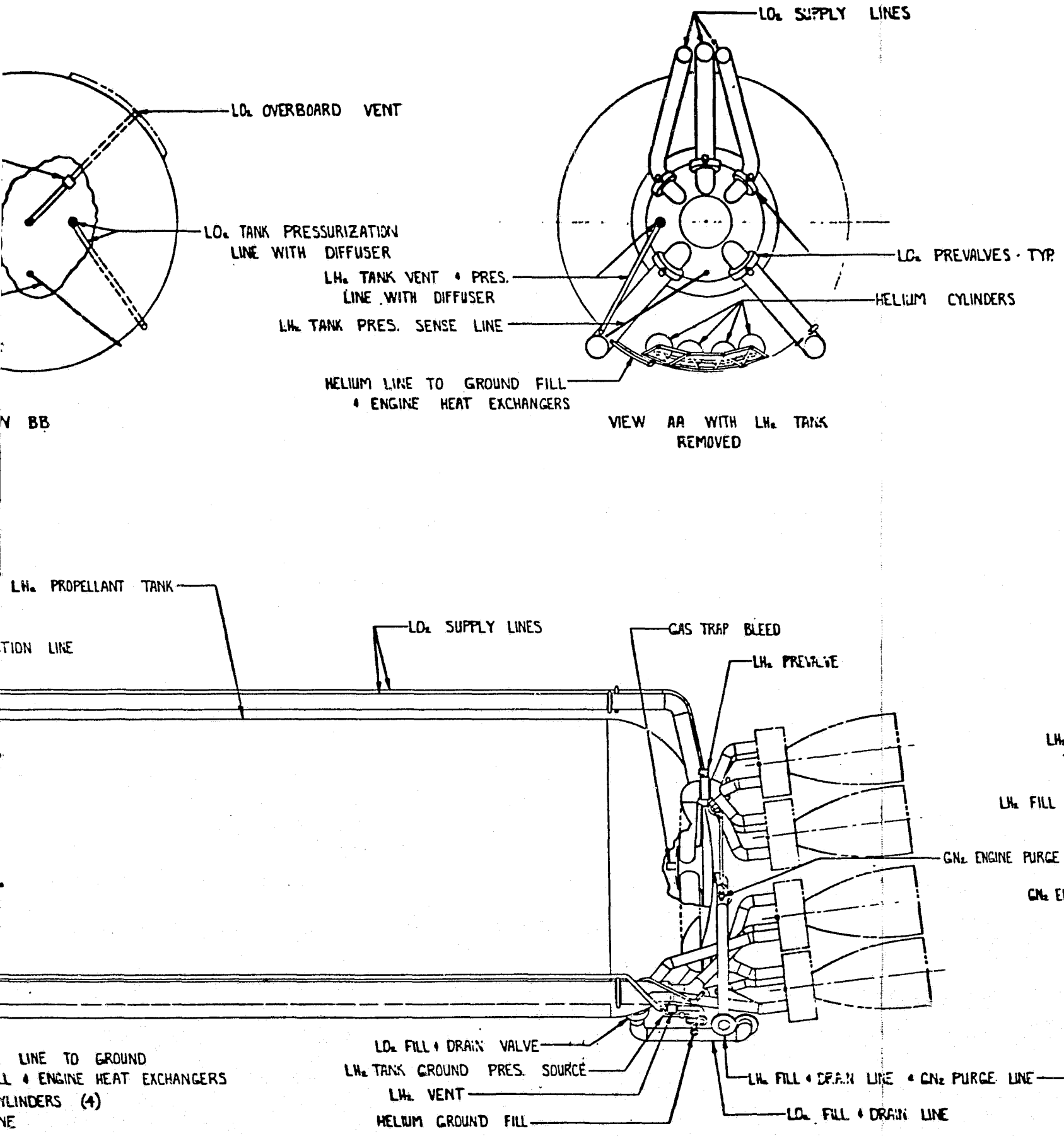
For the 16-engine design, five main LO_2 supply lines were selected. Prior to the selection, several different line configurations were examined, including 16 separate supply lines both internal and external to the LH_2 tank, five supply lines, and four supply lines. The designs were evaluated on the basis of minimizing line propellant residuals (which meant reducing or eliminating the horizontal line length at the aft end of the vehicle) and allowing engine throttling or shutdown of the outboard engines prior to termination of booster burn. The minimum-residual requirement eliminated propellant lines crossing the vehicle centerline. Structural clearance limitations allowed a 30-in. diameter line to be brought down both sides from the tank. These were large enough to supply all nine engines on the lower half. Single 14-in. diameter lines brought through the same space would supply only six engines.

Toward the end of booster burn, the lateral cg shifts toward the orbiter vehicle and it is desired to shift the thrust line by throttling outboard engines to minimize gimbal angles. This is accomplished by providing three lines for the upper seven engines. The four outboard engines are throttled to shut down completely and the remaining three center engines throttled as required, maintaining maximum allowable vehicle acceleration, with the lower nine engines running at full thrust. Thermal protection of the three external lines is provided. For a 15-engine design, these three lines would be combined into two lines.

Individual propellant lines to each engine are desirable from a development testing viewpoint if all the lines are dynamically similar. This would permit the bulk of propellant feed subsystem development testing to be performed with one engine on one set of ducting. This similarly may be obtained if the lines are routed through the LH_2 tank. There are problems inherent in such an approach, including heat transfer, leak detection, maintenance, and reliability that warrant further study. Because such a study has not been made, this approach was not taken for the final study vehicles.



FOLDOUT FRAME 1



FOLDOUT FRAME 2

Figure 2-48. FR-3 Boos

:S

LC₂ PREVALVES - TYP.

.IUM CYLINDERS

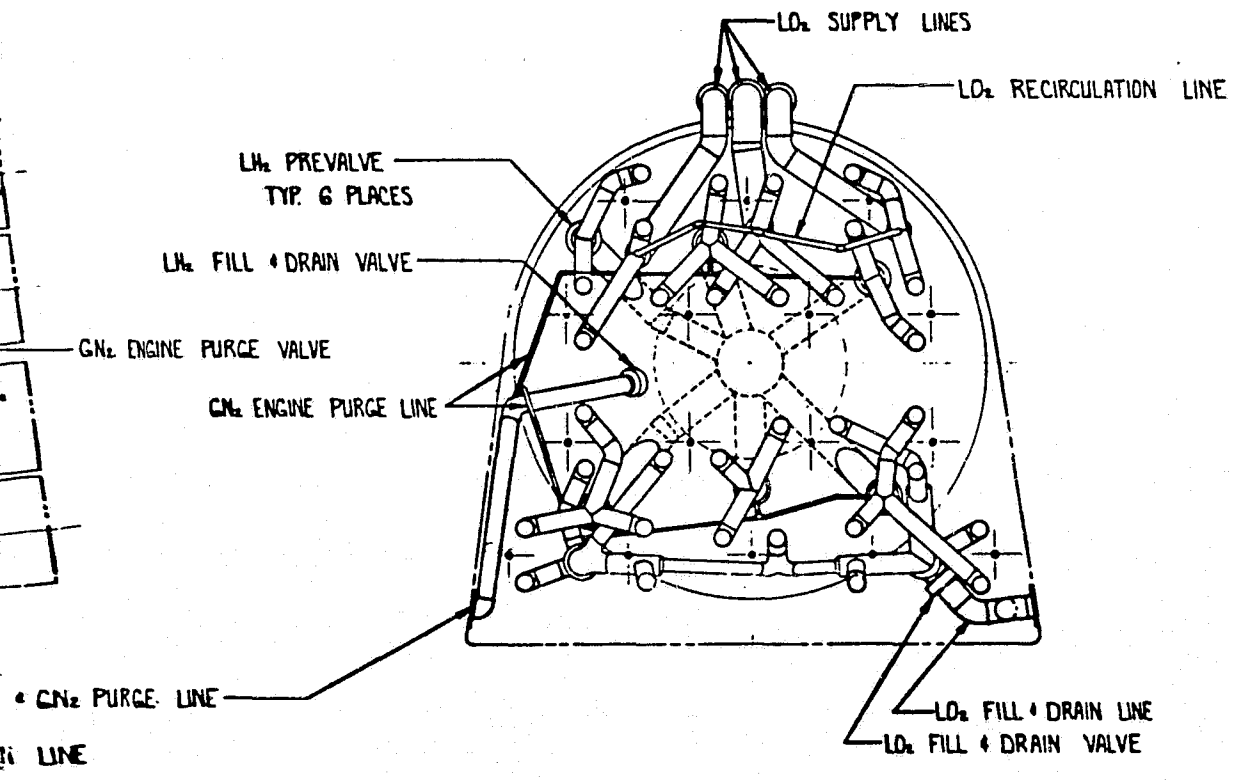


Figure 2-48. FR-3 Booster Propulsion Installation

FOLDOUT FRAME 3

Bellows in the LO_2 ducting provide for required motion due to thermal and structural effects. Multi-ply double-wall bellows are used to reduce cycle failure due to flow-induced vibration and to provide safety should one wall fail. Duct prevalves provide for servicing and checkout, and pressure-relief check valves provide insurance against duct overpressurization failure during engine checkout.

A system integral with the engine (Figure 2-49) provides the required gimbal motion.

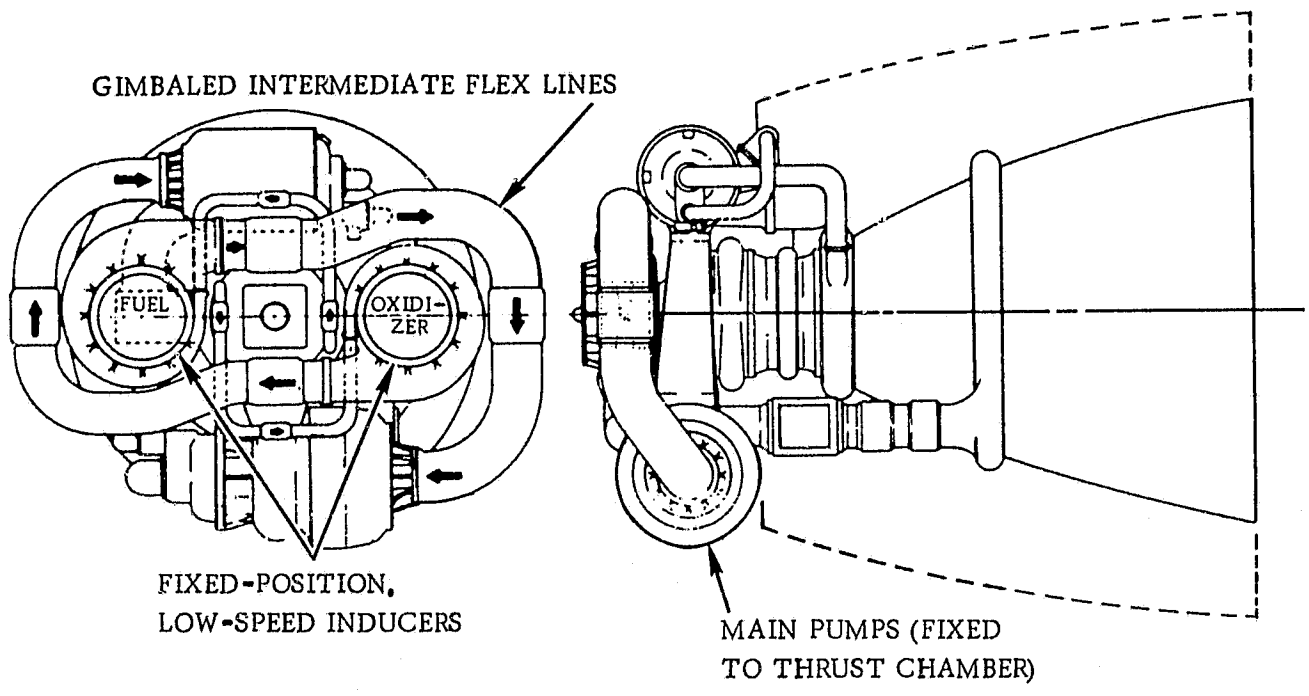


Figure 2-49. Engine Gimbal Provisions

The low-speed inducers (pump inlets) are fixed in relation to the vehicle structure. Intermediate pressure lines between the low-speed inducer and the second-stage pumps (fixed to thrust chamber) are provided with flex joints.

Overheating and geysering of the LO_2 in the long longitudinal (vertical) feed lines is prevented by utilizing natural thermal recirculation of LO_2 through the multiple vertical duct system. Recirculation lines connect appropriate lines at the manifold inlets to provide the recirculation path.

During engine start, the engine pumps accelerate rapidly, incurring additional (inertia) pressure losses above the nominal design friction losses. With the lines sized so that flow velocity throughout is equal to the engine pump inlet flow velocity, the required 16 ft NPSH is satisfied throughout the start transient, without special provisions (added tank pressure or, alternately, propellant subcooling).

With the LO_2 tanks placed forward, the LO_2 in the main longitudinal ducting must be used to achieve low residuals (see Figure 2-50). Following tank depletion, engine cutoff is initiated when the liquid level in the duct falls to the minimum LO_2 NPSH

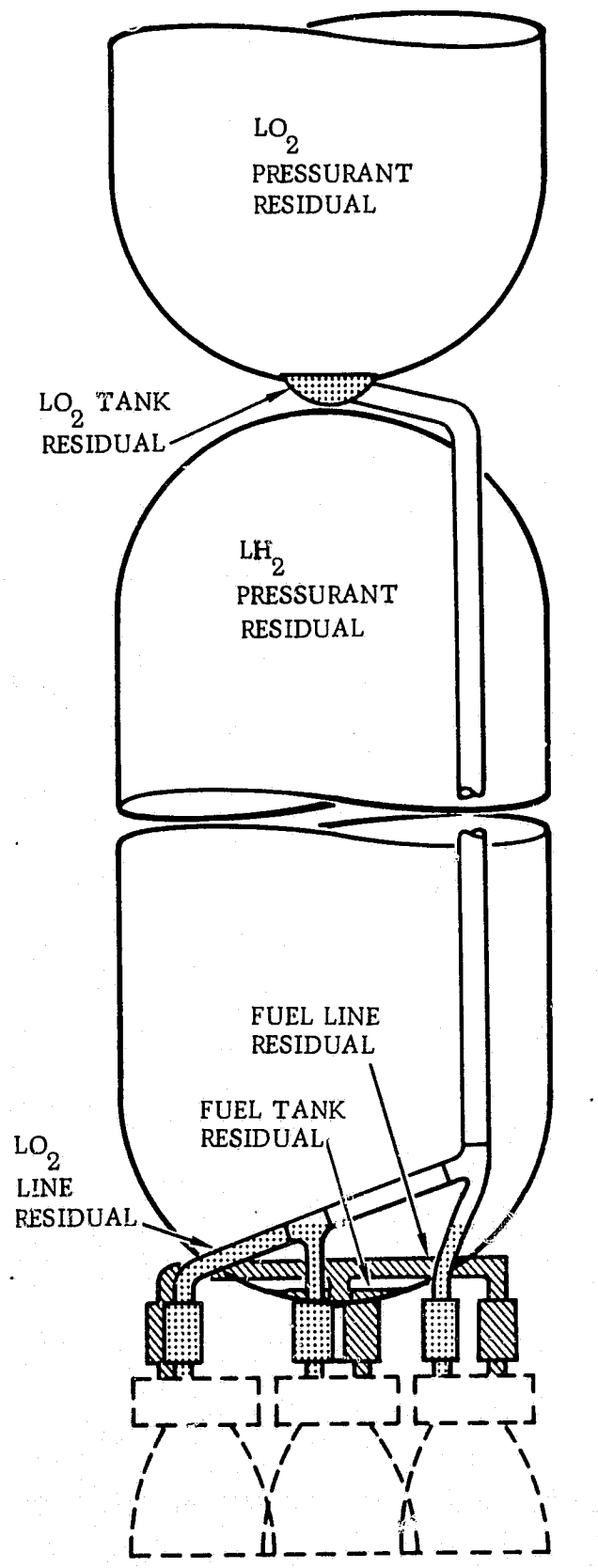


Figure 2-50. Residuals

(before gas entrainment into the engine). Throttling to 20% is used to reduce the NPSH requirement, line friction losses, and resulting residuals.

With the long LO_2 lines, pressure surges at engine shutdown can be quite high for LO_2 valve-closure times approaching the time required for a pressure wave to travel from the valve to the tank and back again. To limit surge pressures to values below the design proof pressure of the ducting, valve closure times of 1.5 seconds or longer are required.

The tank is pressurized to prevent two-phase flow into the tank outlet and to minimize development of a large stratified or boiling layer at the top of the tank. The hydrostatic head in the long propellant lines provides the required NPSH at the pump. Helium is used as pressurant to minimize unusable pressurant residual weight. The helium is stored at 1000 psia in bottles within the LH_2 tank, and is heated to 500°R in an engine-mounted heat exchanger before use. The estimated tank pressure schedule is shown in Figure 2-51.

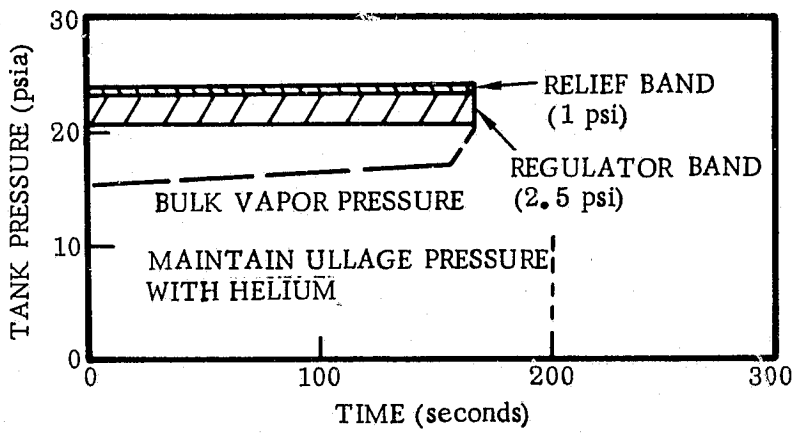


Figure 2-51. FR-3 LO_2 Tank Pressure Requirements (Booster)

The LO_2 tank is uninsulated but protected from wind and moisture condensation by a dry nitrogen purge in the space between the entry heat-shield and the tank. Prior to launch the tank is pressurized to about 20 psia from a ground helium supply.

Filling and draining is accomplished through a connection to one of the main propellant feed ducts, as shown in Figure 2-48. Topping (replenishment of boiloff) is also accomplished through this duct until two minutes prior to engine start, at which time the fill-and-drain valve is closed and the tank is pressurized. The fill line below the fill-and-drain valve is drained of residual propellants before launch, and the fill-and-drain connection is retracted during liftoff. Chilldown gases from the ground fill lines are vented through an overboard facility vent. Gases generated during vehicle chilldown are vented through the vehicle boiloff/vent valve into the atmosphere.

2.4.1.1.2 Liquid Hydrogen (Fuel) Subsystem. Aft positioning of the fuel tank, resultant short lines, and relatively low operating pressures establish the design of the fuel system. Fuel is withdrawn from a central sump under a flat-plate baffle which prevents vortexing and minimizes residuals at depletion. Ducts are sized for pump-inlet flow velocities. Internal manifolding (inverted outlet duct) allows the engines to be mounted near the tank bottom, eliminates the need for insulating the portion of the ducting inside the tank, and simplifies routing of the LO_2 manifold ducting. Multiply double-wall bellows and double-wall external ducting are used to reduce cycle

failure due to flow-induced vibration, to provide safety should one wall fail, and to prevent air liquifaction. Gimbaling flexibility is provided within the engine as shown in Figure 2-49. Duct prevalves provide for servicing and checkout, and pressure-relief check valves provide insurance against duct overpressurization failure during engine checkout.

Because of the short LH_2 lines, no geysering problem is expected. A bleed is provided for the high point (trap) in the inverted fuel outlet duct, preventing formation of a gas pocket during no-flow conditions.

The propellant will reach saturated conditions in the ducting during tanking. After pressurization for engine start, the added heating and resulting vapor pressure rise is small compared to the added tank ullage pressure of 24 psi.

The short-coupled fuel system with large lines has very low transient loss during start. The 60-ft NPSH requirement and line friction pressure loss are satisfied by the engine prestart pressure requirement of 40 psia.

The short-coupled lines and high flow velocities require that residuals be determined at the time pull-through into the tank outlet is initiated in order to prevent gas entrainment into the engines. Shortly before tank depletion, the 60-ft NPSH requirement is satisfied by tank pressure. At tank depletion, the engines are shut down from 20% thrust and the liquid pull-through level is minimized by the outlet-baffle design, resulting in low residuals.

With valve closure time regulated to greater than 1.5 seconds for the LO_2 system, there is no expected pressure-surge problem in the fuel system.

Autogenous H_2 bleed pressurant at 300°R is provided by the main propulsion system to supply the necessary pump NPSH. A pressure of 26.3 psia is required prior to burnout. The estimated tank pressure scheduled is shown in Figure 2-52. The tank is insulated with either fiber-reinforced 3D foam (0.62 lb/ft^2) or open-cell insulation (0.78 lb/ft^2).

The tank will be pressurized prior to launch with ground helium to provide the required prestart pressure of 40 psia.

Filling and draining are accomplished through a duct into the bottom of the fuel tank, separate from the engine fuel-inlet ducts. Operation is similar to the LO_2 system previously described, except that all chilldown gases are vented through the vehicle boiloff/vent valve into the facility vent stack for disposal. A ground helium purge supply is also provided prior to tanking and in case of launch abort.

2.4.1.2 Orbiter Propellant Feed. Figure 2-53 is a detail layout of the FR-3 orbiter design, which is similar to the FR-3 booster just described, except as specifically discussed in the following paragraphs.

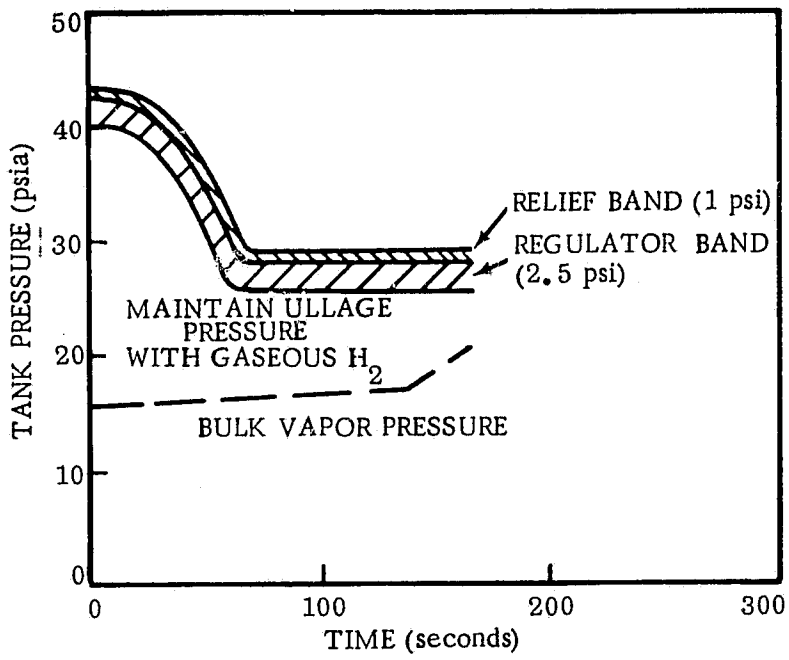


Figure 2-52. FR-3 LH_2 Tank Pressure Requirements (Booster)

2.4.1.2.1 Liquid Oxygen Subsystem. An inverted LO_2 tank outlet line is used to prevent intrusion of ducts into the payload area. An increase in tank pressure of 2-3 psi (equal to the liquid head in the inverted duct above the tank liquid level, times the vehicle acceleration, plus the baffle and line pressure drop) is required to prevent cavitation in the outlet system. The tradeoff of this added tank and pressurant weight versus the effect of a longer vehicle will determine the final selection of inverted vs. conventional (down inlet) design. This tradeoff has not yet been performed.

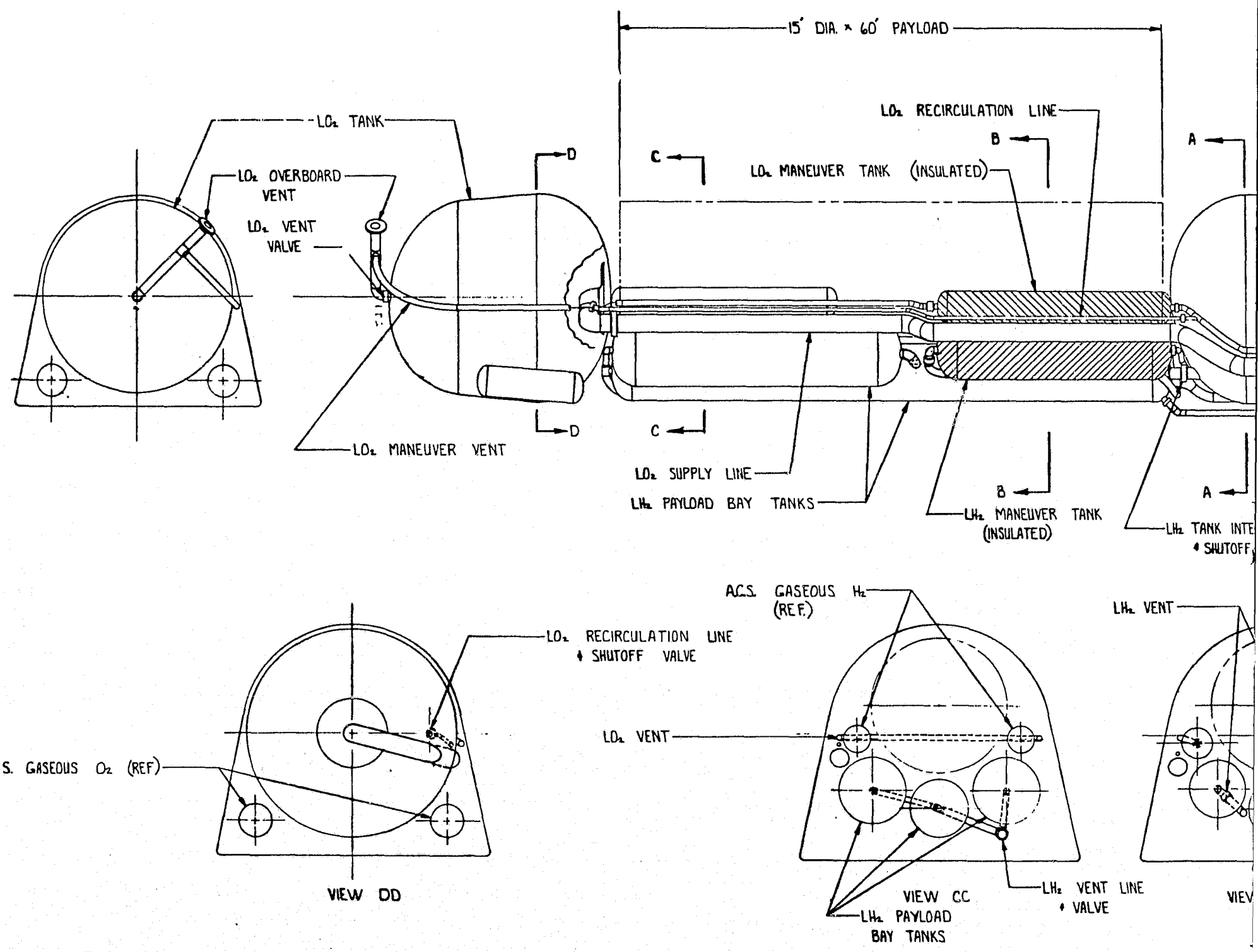
Dual ducts are provided. One line is sized for nominal flow for the three engines. The other line is sized for orbit maneuver propulsion operation (one engine at 20%) and connects to both the main tank (thus providing a prestart recirculation flow path) and the orbit maneuver tank. Isolation valves are provided for orbit maneuver phase.

The orbiter engines are started before booster cutoff, while sufficient hydrostatic head is available to meet the NPSH requirement of 16 ft during the start transient.

Autogenous O_2 bleed from the engine is used since the pressurant residual is used for ACS propellant. The tank is pressurized to about 20 psia with ground helium prior to launch and is vented as required during the booster phase to maintain 20 psia. The estimated tank pressure schedule is shown in Figure 2-54. After burnout, the tanks are maintained at 25 psia. Residuals and boiloff are used to pressurize the maneuvering tanks and to operate the ACS engines.

2.4.1.2.2 Liquid Hydrogen (Fuel) Subsystem. The fuel tank remains at 40 psia during boost phase in order to provide the required prestart pressure. Figure 2-55 shows the estimated tank pressure schedule (29.3 psia required prior to burnout).

2.4.1.3 Orbit Maneuver Propellant Subsystem. Separate, highly insulated tanks located in the payload bay provide propellant for the four orbit-maneuver firings of the main engines. The tanks are sized for approximately 39,000 lb of propellant at MR = 5:1. The orbit-maneuver tanks are connected to the main propellant manifolds by separate lines sized to provide flow for 20% thrust of one engine. This sizing is predicated on the use of orbit-maneuver propellants to provide additional velocity required for once-around abort with one orbiter engine out. During orbit-maneuver firings, maximum engine thrust is 10% of nominal (one engine only). Between each firing,



FOLDOUT FRAME 1

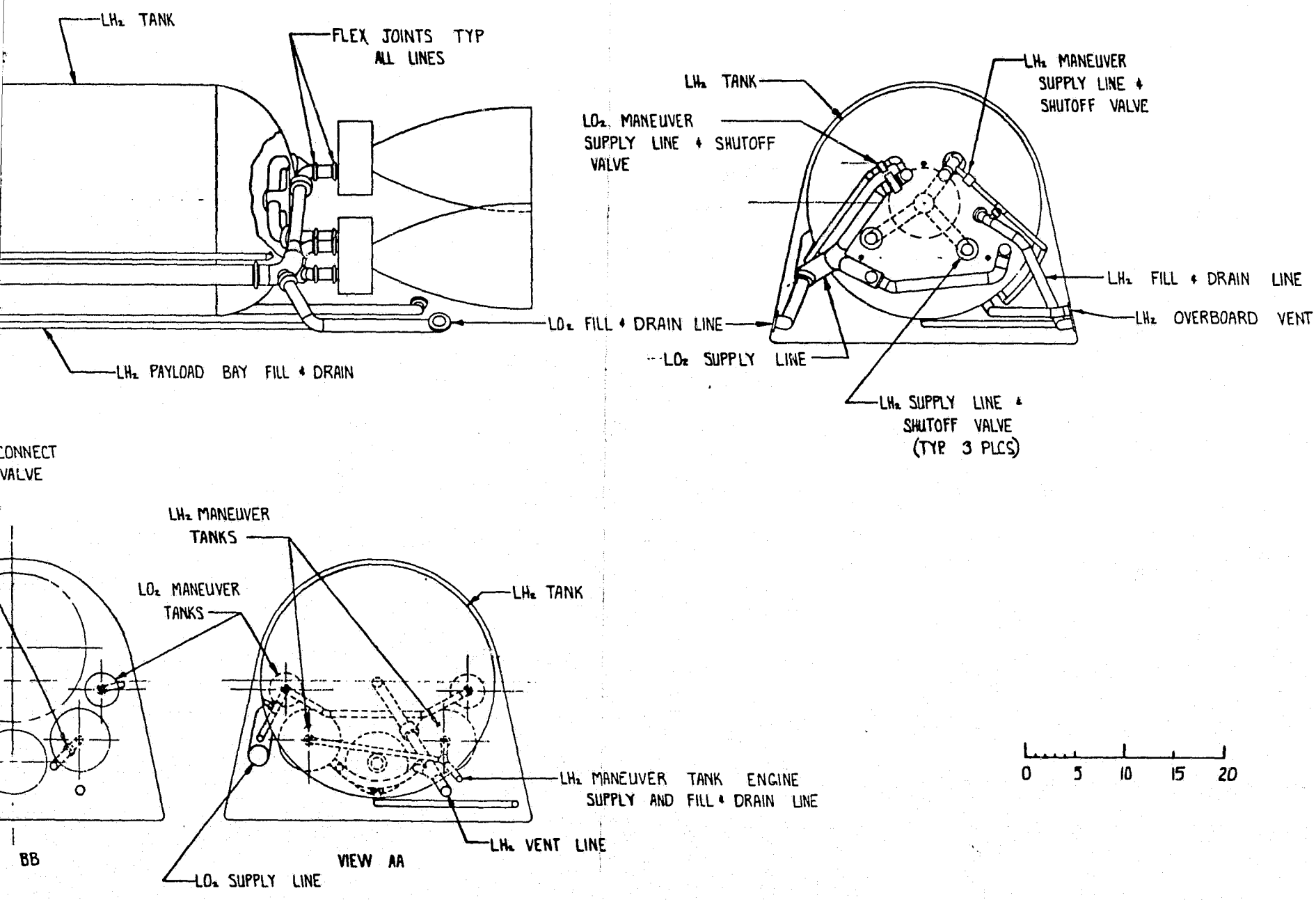


Figure 2-53. FR-3 Orbiter Propulsion Installation

FOLDOUT FRAME 2

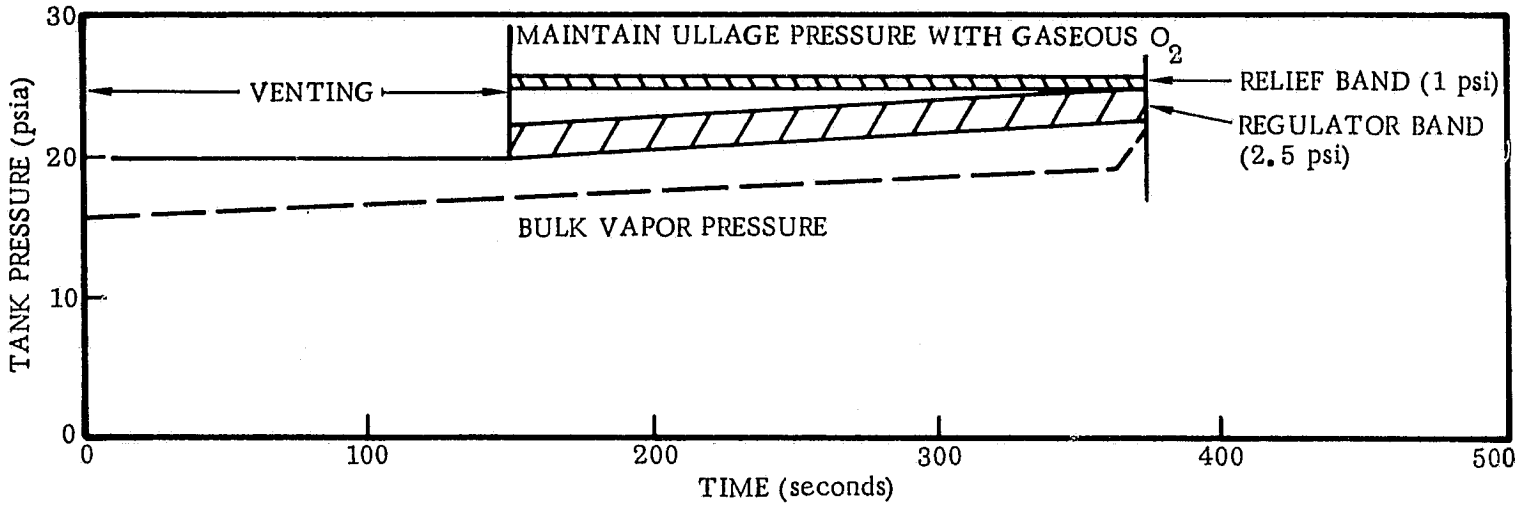


Figure 2-54. FR-3 Orbiter LO_2 Tank Pressure Requirements

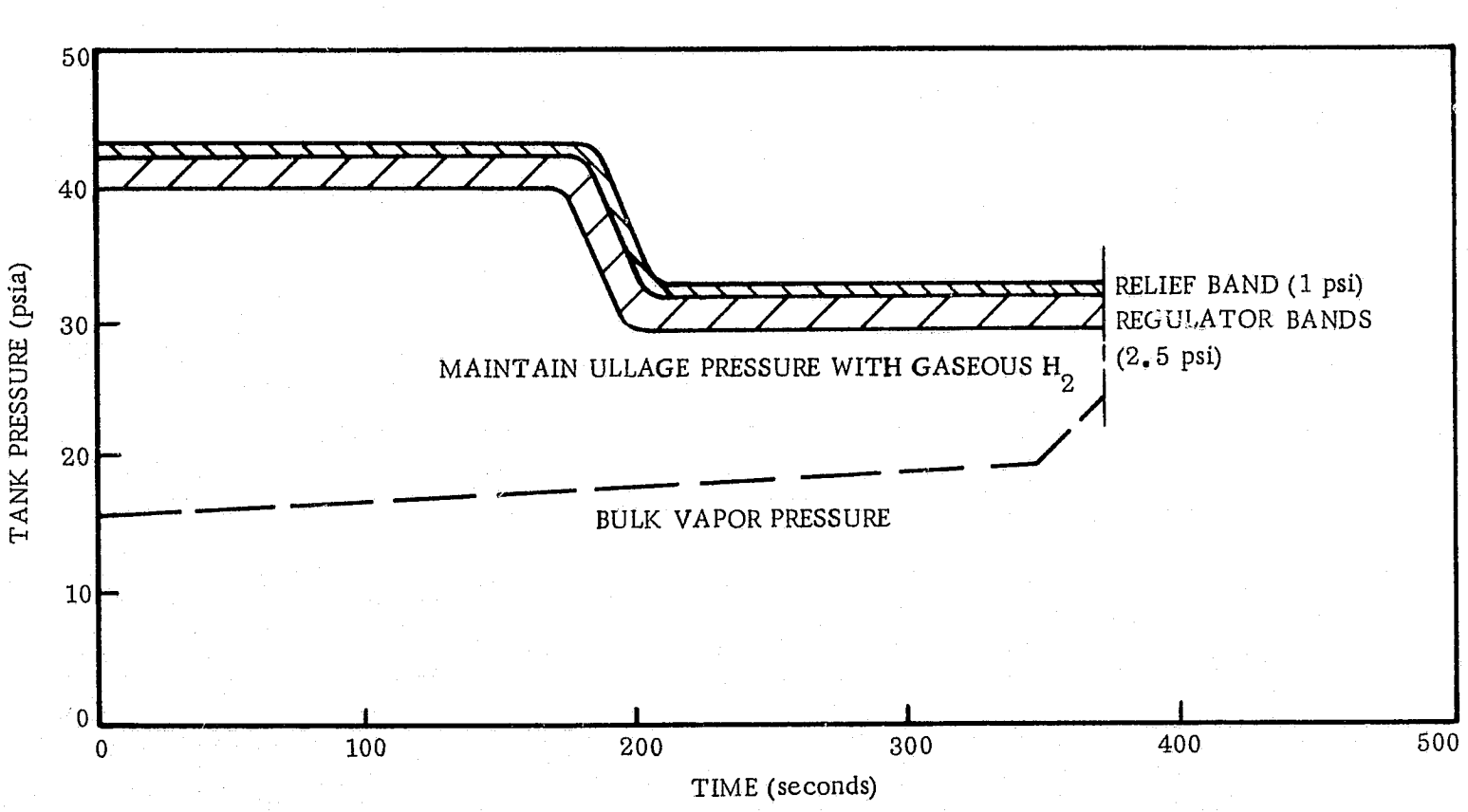


Figure 2-55. FR-3 Orbiter LH_2 Tank Pressure Requirements

residual line propellants are returned to the tanks to eliminate propellant boiloff loss which would be incurred if the main engine were kept wet (heat soak-back and solar input). Attitude control propulsion combined with helium pressurization is used to empty the lines of propellants. Tank pressurization is provided from the main tanks. (A zero-g vent on the hydrogen maneuver tank is provided, with the ventage routed to cool the LO_2 maneuver tank to prevent LO_2 boiloff.)

The LO_2 maneuver tanks were located in the aft portion of the payload bay to utilize existing available space. It is desirable to minimize the distance between the orbit maneuver tanks and the engines. By locating the engines further aft relative to the main tank, it may be possible to close-couple the tanks to the engines, thus reducing line and insulation weight, cooldown and boiloff propellant losses, and start transient effects. Moving the engines aft adds structural weight and moves the vehicle cg further aft.

2.4.2 ATTITUDE CONTROL. The attitude control propulsion subsystem consists of 48 attitude control thrusters with nominal thrusts of 2500 pounds. The thrusters are supplied with gaseous oxygen and hydrogen from high-pressure accumulators, which are sized to provide the propellant required for entry control. These accumulators may be charged on the ground, with main-engine bleed gas, auxiliary pumps, or compressor. If compressors are used, residual gases and liquids may be used to minimize or eliminate the need for additional propellant for attitude control. The engines may be operated at low pressure directly from the main tanks during certain orbital phases where high thrusts are not required and low impulse bits are desired.

2.4.3 AIRBREATHING (FLYBACK) ENGINES. The FR-3 space shuttle vehicle has airbreathing flyback engines in the booster element and the orbiter element. Booster engines will be deployed at an altitude of 25,000 feet or higher and windmill-started during glide to a normal cruise altitude of about 15,000 feet. The booster element will then fly back 250 to 300 n.mi. and land at the landing site. Orbiter engines will be deployed at an altitude of 15,000 feet or higher and windmill-started during glide toward the landing site. The engines will then operate at idle to an altitude of about 1500 feet when they will be brought to the thrust required for a powered approach and landing.

Two final FR-3 vehicle designs are being presented: one with a 15-ft-diameter payload bay and one with a 22-ft-diameter payload bay. For the final designs actual flyback engines were chosen and the airbreathing subsystem weights in the final synthesis computer runs reflect the installation of these engines.

The FR-3 is a two-element vehicle with the booster element having a flyback weight approximately 60 percent heavier than the FR-4 boosters, and the orbiter element having a flyback weight about 10 percent lighter than the FR-4 orbiter.

The booster element requires more and larger engines than the FR-4 booster while the size of the 40,000-pound high-bypass turbofan engines creates installation problems in the orbiter element.

Preliminary analysis of the booster element indicated a maximum sea level static thrust (MSLST) requirement of greater than 200,000 pounds. The 40,000-pound class candidate engines listed in Table 4-1 have growth versions in the 50,000-pound class

that are projected for manufacture. A four-engine installation of one of these growth version engines appears the most reasonable.

The Rolls Royce RB211-56 engine was selected for the booster final design. It has a 52,500-lb MSLST. The installed thrust-to-weight ratio was assumed to be the same as for the RB211-22. The engine performance used to evaluate the flyback capabilities of the final FR-3 designs is the RB211-56 specification (Reference 4-4). Ten percent intake recovery and power offtake losses were assumed. FR-3 final synthesis computer run parameters used in evaluating the booster flyback performance are:

	15-ft-dia Payload	22-ft-dia Payload
Flyback weight, lb	564,378	585,927
Flyback range, n.mi.	281	271
Planform area, ft^2	8,170	8,430
Max L/D	7.2	7.2
C_L at Max L/D	0.45	0.45

The flyback conditions, performance requirements, and capabilities of the FR-3 boosters under normal conditions are:

	15-ft-dia Payload	22-ft-dia Payload
Cruise Altitude, ft	15,000	15,000
Cruise Velocity, knots	269	269
Required Cruise Thrust, lb	78,386	81,379
Available Thrust at Max Cruise Rating, lb	79,625	79,625
Available Thrust at Emergency Max Cont., lb	85,586	85,586

Thrust available from the four turbofan engines at maximum cruise rating meets the requirements of the 15-ft-diameter payload vehicle but is not sufficient for the 22-ft-diameter payload vehicle. It will be necessary, therefore, for the 22-ft-diameter payload vehicle to fly at a slightly lower altitude or for the engines to operate between the maximum cruise rating and the emergency maximum continuous rate.

The booster is required to have the capability to fly back to the landing site with one engine inoperative. Comparisons of final performance and synthesis run values are:

Engine Out Conditions	15-ft-dia Payload	22-ft-dia Payload
Cruise Altitude, ft	7,900	6,400
Cruise Velocity, knots	239	234
Synthesis Input Velocity, knots	243	243
Required Fuel, lb	47,169	47,787
Synthesis Run Fuel, lb	46,916	47,035
Fuel Shortage, lb	253	752

The fuel available in the synthesis run is slightly less than that required to fly back. However, there is 9,000 pounds of ballast provided in the nose of both configurations for balance during hypersonic entry. This is not required during subsonic cruise, so a portion of the ballast weight could be converted to flyback fuel to provide the required fuel plus a reserve (in case of headwinds). This would neither impose a weight penalty nor compromise the stability of the vehicle.

Preliminary analysis of the orbiter element flyback requirements indicated a total MSLST requirement of about 60,000 pounds. The Pratt and Whitney TF33-P-7 turbofan engine has a maximum sea level static rating of 21,000 pounds, so three of these engines would match the requirements. This older engine has a bare thrust-to-weight ratio of about 4.5 and so would weigh more than an advanced turbofan engine of the same thrust rating but there are no advanced turbofan engines in this thrust range under development. An alternate selection would be two of the 40,000-pound thrust advanced engines. The installation weight would be approximately the same; however, the size of these larger engines would cause installation difficulties. The three-engine TF33-P-7 configuration was chosen for the final design. Synthesis computer run parameters and climb and go-around capabilities of the two FR-3 orbiter final designs are:

	15-ft-dia Payload	22-ft-dia Payload
Flyback weight, lb	289,655	319,207
Planform area, ft^2	4,910	5,416
Max L/D	7.8	7.8
C_L at Max L/D	0.55	0.55
Cruise velocity, knots	178	178
Required cruise thrust, lb	37,135	40,923
Max available thrust, lb	47,520	47,520
Climb capability: Rate, ft/min	645	372
Angle, deg	2.1	1.2

In the 15-ft-diameter payload FR-3 design the four 52,500-pound engines in the booster and three 21,000-pound engines in the orbiter provide satisfactory flyback and landing capability. In the 22-ft-diameter payload FR-3 design, the performance requirements are met but the capability is marginal.

The engines use JP-4 fuel for flyback and landing. Hydrogen is being considered as a replacement, however. Section 4.6 describes the potential weight saving to be gained by using hydrogen in the FR-3 flyback engines.

2.5 FR-4 TWO-STAGE SEQUENTIAL BURN VEHICLE

FR-4 propulsion subsystems and their modes of operation are similar to the ones in the FR-3.

2.5.1 MAIN PROPULSION. Each booster element in the FR-4 has nine engines instead of the 15 engines of the FR-3. The 400K-lb-thrust engines are identical to the engines used in the FR-3. The FR-4 orbiter propellant feed subsystem is essentially identical to the FR-3 orbiter just described.

The FR-4 booster (two elements) is similar to the FR-3 booster except for fewer engines on each element, resulting in simpler propellant feed design. No detailed design layout drawings were made.

2.5.2 ATTITUDE CONTROL. The attitude control propulsion subsystem consists of forty-eight 3500-pound thrusters with associated propellant subsystems.

2.5.3 AIRBREATHING (FLYBACK) ENGINES. As with the FR-3 vehicle, two final FR-4 vehicle designs are being presented. One has a 15-ft diameter by 60-ft long payload bay. The other has a 22-ft by 60-ft payload bay. For these final designs an actual engine that will be available was chosen. Airbreathing subsystem weights in the final synthesis computer runs reflect the installation of this engine.

Preliminary analysis of the FR-4 vehicle indicated that total maximum sea level static thrust required for the booster element to cruise back at 15,000 ft altitude is in the 120,000-pound range. Installation studies discussed in Section 4.5 concluded that a three-engine configuration is preferred if engines are available with thrusts that match requirements. Several of the advanced turbofan engines listed as candidate engines are in the 40,000-pound class. The Rolls Royce RB211-22 engine has a slightly better thrust-to-weight ratio than do the other engines and was therefore selected as the engine for the FR-4 final design. Figure 4-8 shows an installation arrangement of this engine. Bare engine weight of the RB211-22 is 6353 pounds. Installation weights including cowls, inlet duct, tailpipes, etc., were estimated to be 1293 pounds, resulting in a total weight per engine of 7646 pounds.

The final airbreathing propulsion subsystem configuration for the FR-4 vehicle is three RB211-22 engines on each booster element and two RB211-22 engines on the orbiter element. Engine performance values used to evaluate flyback capabilities of the final FR-4 designs are from the RB211-22 specification (Reference 4-3). Estimated performance values listed in the specification are for 100% intake recovery and no offtake losses. To account for these losses, the values were decreased by 10% in making the evaluation. FR-4 final synthesis computer run parameters used in the evaluation are:

	15-ft-dia Payload	22-ft-dia Payload
Booster Flyback Weight, lb	355,500	390,288
Orbiter Flyback Weight, lb	325,700	341,478
Booster Flyback Range, n.mi.	255	276
Booster Planform Area, ft^2	6,072	6,603
Orbiter Planform Area, ft^2	5,565	5,955
Max L/D	7.8	7.8
C_L at Max L/D	0.55	0.55

Flyback conditions, performance requirements, and capabilities of the FR-4 boosters are:

Normal Conditions	15-ft-dia Payload	22-ft-dia Payload
Cruise Altitude, ft	15,000	15,000
Cruise Velocity, knots	224	224
Required Cruise Thrust, lb	45,577	50,037
Available Thrust at Max Cruise, lb	51,127	51,127

Thrust available from the three turbofan engines at maximum cruise rating is greater than that required for both the 15-ft-diameter and the 22-ft-diameter payload vehicles. Therefore, the booster flyback engines can operate at partial power or the booster could cruise at a higher altitude and velocity if this is desirable.

The booster is required to have the capability to fly back to the landing site with one engine inoperative. Since in both designs the engines provide more than enough thrust normally to fly at 15,000 feet, the booster can fly with one engine out at a higher altitude and velocity than if the engines furnished just enough thrust under normal conditions. Comparisons of final performance and synthesis run values are:

Engine-Out Conditions	15-ft-dia Payload	22-ft-dia Payload
Cruise Altitude, ft	7,300	3,800
Cruise Velocity, knots	197	188
Synthesis Input Velocity, knots	178	178
Required Fuel, lb	28,543	35,061
Synthesis Run Fuel, lb	30,711	36,399
Reserve Fuel Available, lb	2,168	1,338

Some reserve fuel for booster flyback is available. Additional fuel would be needed if strong head winds were encountered. In both FR-4 designs, 11,000 pounds of ballast are required in the nose of the booster to provide the proper balance during hypersonic entry, but is not required during subsonic cruise. A portion of this ballast could be converted to flyback fuel for a reserve in case of headwinds, therefore, without either increasing the weight of the vehicle or compromising its stability.

The two turbofan engines on the orbiter element are to provide the thrust for a powered approach and landing with the additional capability to climb, go-around, and make a second approach and landing. Performance capabilities of the two FR-4 designs are:

	15-ft-dia Payload	22-ft-dia Payload
Cruise Velocity, knots	178	175
Required Cruise Thrust, lb	41,756	43,779
Available Max Takeoff Thrust, lb	56,585	56,765
Climb Capability		
Rate, ft/min	817	675
Angle, deg	2.6	2.2

In both FR-4 designs, the three 40,600-pound turbofan engines on the booster element provide more than enough thrust to meet the flyback requirements. The two engines on the orbiter element provide thrust for approach and landing as well as limited climb capability for go-around if necessary.

2.6 REFERENCES

2-1 Preliminary Tradeoffs and Requirements for Boost, Maneuver, and Reaction Control Propulsion Systems, Convair division of General Dynamics, Report GDC-DCB69-024, 29 July 1969.

2-2 Rocketdyne Shuttle Vehicle Propulsion, Rocketdyne Division of North American Rockwell Corporation, Report No. BCI 69-72, August 1969 (Confidential, title unclassified).

SECTION 3

ATTITUDE CONTROL PROPULSION SUBSYSTEM

The attitude control propulsion subsystem provides orbiter attitude control during orbital and entry phases of flight, minor velocity increments for rendezvous and docking, drag makeup, propellant settling, and roll control during orbit maneuver firings. The major study effort in attitude control propulsion has been to identify thruster sizes and arrangements to meet the vehicle control requirements, to determine thruster duty cycles, and in the definition of a gaseous O_2/H_2 propellant subsystem to satisfy the mission requirements.

3.1 REQUIREMENTS

The attitude control propulsion subsystem (ACPS) for the space shuttle may be required for the following functions:

a. Attitude control during the orbital and entry phases of flight.

b. Drag makeup.

c. Rendezvous and docking.

d. Propellant settling.

e. Roll control during the orbit maneuver firings.

3.1.1 MISSION PROFILES. Figures 3-1 and 3-2 show typical mission profiles and requirements for the space shuttle ACPS. For the low earth-orbit mission, propellant equivalent to an on-orbit ΔV of 15 fps* is required for payload deployment and retrieval and for drag makeup during seven days in low earth orbit. At the end of the seven-day mission, an equivalent ΔV of 50 fps is required for attitude control during entry. The second mission is a space station logistics mission. After orbiter main engine shutdown, the vehicle coasts from an altitude of approximately 45 n.mi. to an altitude of 100 n.mi. During this time the ACPS provides coarse attitude control. After circularization, the vehicle may stay in the 100 n.mi. orbit for up to 18 hours for phasing purposes. During this time, coarse vehicle attitude control is provided. Any time between the end of the circularization burn and 18 hours, the vehicle may transfer to a higher orbit. During this transfer, the ACPS provides coarse attitude control. When the vehicle reaches approximately 260 n.mi., approximately 10 n.mi. below the target orbit, the orbit is circularized again using the main engine. Closure with the space station is performed with the ACPS, with accelerations between 0.03 g and 0.05 g. During the closure maneuver firings, the vehicle attitude is controlled to ± 0.5 deg.

*More recent analyses indicate this requirement may increase to 45 fps.

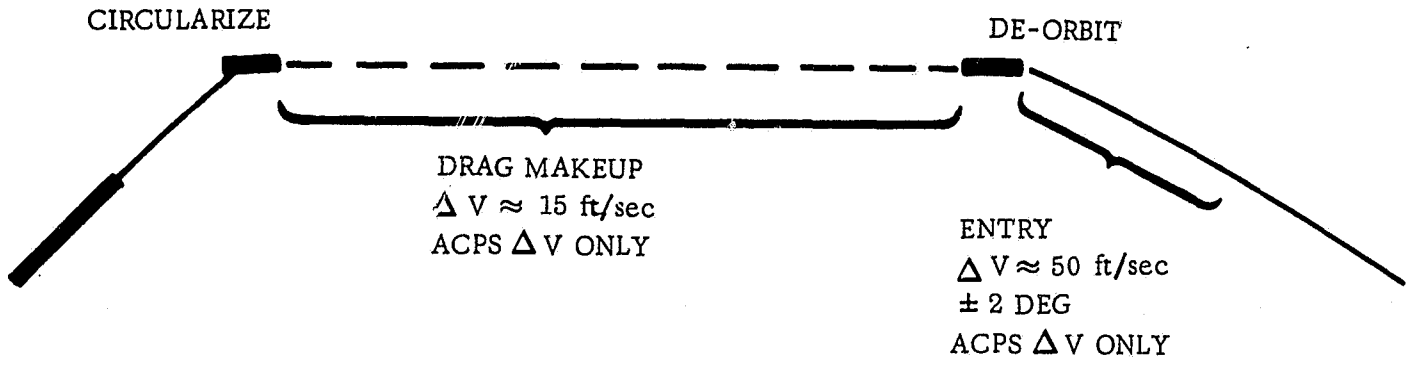


Figure 3-1. Typical Low Earth-Orbit Mission Profile

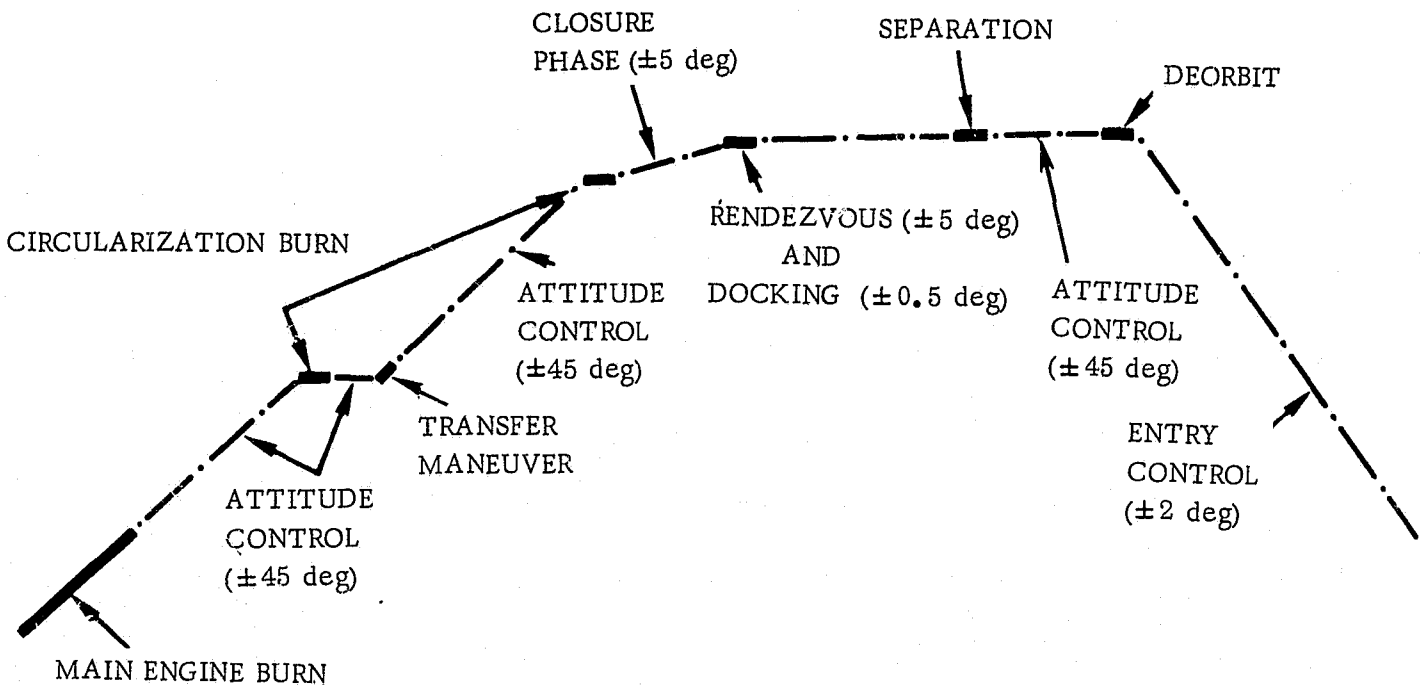


Figure 3-2, ACPS ΔV Requirements

Delta V's of the order of 1 fps are applied during the closure, rendezvous, and docking maneuvers.

No propulsive functions are provided by the space shuttle during the time it is attached to the space station or space base, which may be up to six days.

The space shuttle maneuvers away from the space station, with fine attitude control (±0.5 deg) and ΔVs of one fps, until the space shuttle is over 100 feet away from the target. The main engine is then fired to deorbit the vehicle.

During the entry phase, attitude control rates are 2.5 deg/sec^2 and the vehicle attitude is maintained within ±2 deg of the required angle of attack. As aerodynamic control becomes increasingly effective, the attitude control thruster levels may be decreased.

3.1.2 MANEUVER REQUIREMENTS. The two primary maneuver requirements are (1) rendezvous and docking and (2) entry.

3.1.2.1 Rendezvous and Docking Acceleration Requirements. Figure 3-3 shows the region of linear and angular acceleration required for rendezvous and docking as defined in References 3-1 and 3-2. Criteria provided by NASA (Reference 3-3) require that rendezvous and docking be accomplished at not less than 0.03 g translational capability, and not more than 0.05 g translational capability. This requirement is equivalent to approximately 0.965 ft/sec^2 minimum and 1.6 ft/sec^2 maximum, well within the band defined by Figure 3-3.

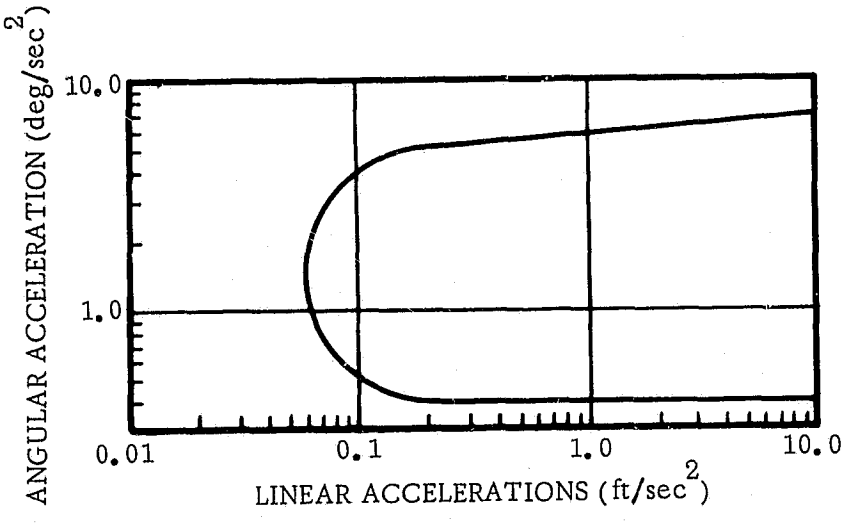


Figure 3-3. Rendezvous and Docking Maneuverability Requirements

3.1.2.2 Entry Requirements. Figure 3-4 shows maneuver requirements for hypervelocity vehicles during the entry portion of flight (Reference 3-4). This defines the acceptable limits for control accelerations in roll, pitch, and yaw, and indicates the selected capabilities of the space shuttle. Data for the X-15 is shown for comparison. The requirement for a minimum of 2.5 deg/sec^2 in pitch, yaw, and roll increased the number of thrusters from 32 to 48 in the final study vehicles. (See p. 3-11.)

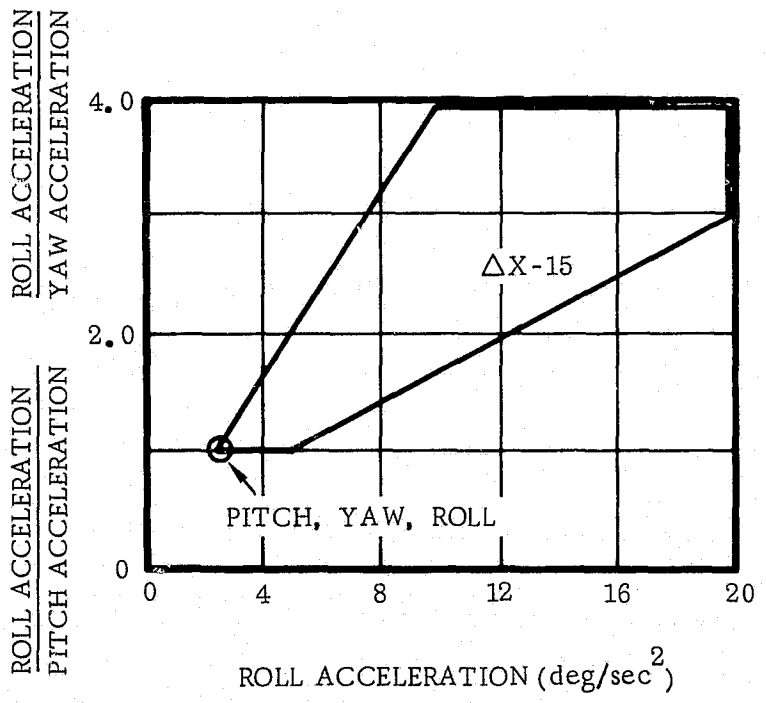


Figure 3-4. Cornell Specification Hypervelocity Vehicle Maneuverability Requirements

Figure 3-5 shows how the ACPS control requirement can be reduced during entry, as the aerodynamic control surfaces become more effective and can provide more of the control. At the left of the curve, the vehicle is passing through an altitude of approximately 400,000 feet at slightly suborbital velocity, and dynamic pressure is negligible. All control must be provided by the ACPS. As entry proceeds, dynamic pressure increases and part of the vehicle control requirements can be assumed by the aerodynamic control surfaces. After 1500 seconds, the full 2.5 deg/sec^2 can be obtained without use of the ACPS.

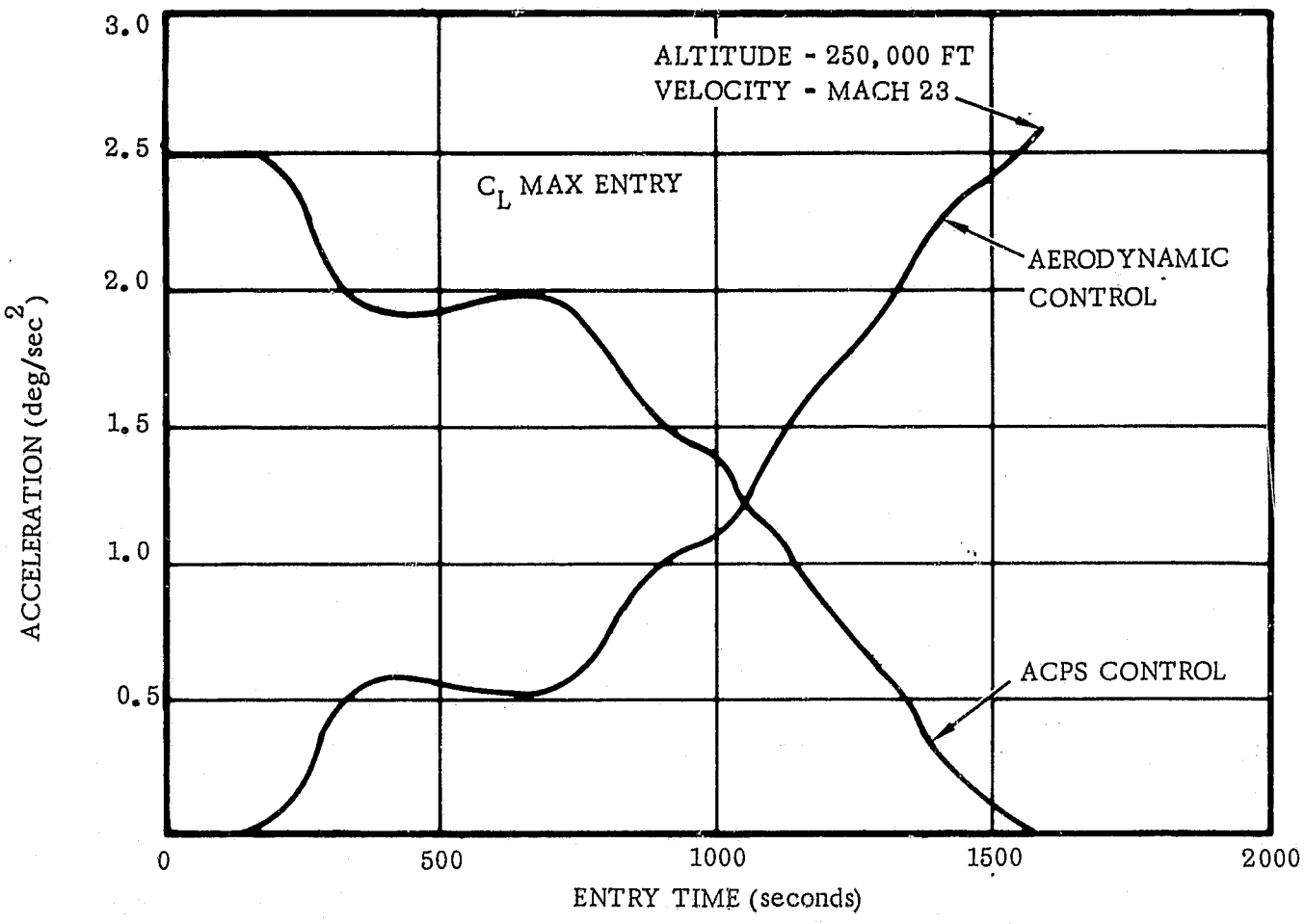


Figure 3-5. Maximum Acceleration vs Reentry Time

At this point, altitude is 250,000 ft and velocity is Mach 23. A tradeoff can be made between the use of ACPS and usage and sizing of the control surfaces. Characteristics shown are typical of orbiters used in this study.

3.1.3 ACPS DUTY CYCLE REQUIREMENTS. Two missions were considered in evaluating the ACPS duty cycle requirements: the low earth orbit mission and the space station logistics mission.

3.1.3.1 Low Earth-Orbit Mission. The low earth-orbit mission is a seven-day 100-n.mi. mission. During the mission the ACPS is required to furnish 15 fps drag makeup and to furnish control during entry. Table 3-1 summarizes the mission duty cycle for the low earth orbit. The drag makeup is accomplished by pulsing two thrusters for the minimum pulse time of 0.25 sec 120 times during the mission in order to furnish 15 fps in drag makeup ΔV.

The duty cycle of the ACPS during entry is based on the requirements shown in Figure 3-5. Angular accelerations of approximately 2.5 deg/sec^2 for 250 seconds, 2 deg/sec^2 for 500 seconds, and a decreasing acceleration averaging 1 deg/sec^2 for 750 seconds result in a ΔV of 60.6 fps. Nominal requirements are probably compatible with stated entry ΔV's of 50 fps.

Table 3-1. Low Earth-Orbit Mission Duty Cycle

Task	Function	% On Time = On-Time × 100 / Mission Time	Interval Per Pulse (sec/pulse)	Number of Engines	ΔV (fps)
Seven-Day Orbit	Drag Makeup	0.000047	5040	2	15
Entry	Control to ±2 deg with 2.5-3 deg/sec^2 for 250 sec	1.87 Roll	13.3	4	
		1.6 Pitch	15.6	8	
		1.6 Yaw	15.6	8	23.1
	Control to ±2 deg with 1.9-2.25 deg/sec^2 for 500 sec	1.4 Roll	17.8	3	
		1.2 Pitch	20.8	6	
		1.2 Yaw	20.8	6	26.0
	Control to ±2 deg with 0.75-1.25 deg/sec^2 for 750 sec	0.47 Roll	53.3	1	
		0.80 Pitch	31.2	4	
		0.60 Yaw	41.6	3	11.5
Total ΔV					75.6

3.1.3.2 Space Station Logistics Mission. The profile for this mission is shown in Figure 3-2. The ACPS furnishes limit cycle attitude control to ±45 deg while in orbit hold or during orbit transfers; orientation to ±5 deg prior to each orbit maneuver burn and during rendezvous; roll control to ±5 deg during each maneuver burn; ΔV to transfer from 260 n.mi. to 270 n.mi. and to rendezvous dock and undock; and orientation to ±0.5 deg during docking and orientation to ± 2 deg during entry. Table 3-2 describes each task, the function performed by the ACPS, the percent of the time that the thrusters are operating, the interval between each pulse, the number of thrusters, and the equivalent ΔV for each maneuver. The total ACPS ΔV for the mission described is 162.1 fps. A typical ΔV used for this mission is 200 fps, which indicates a reserve allowance of 37.9 fps or a tightening of mission requirements.

3.1.4 LOCATION AND GEOMETRY CONSIDERATIONS. Based on available space in the vehicle, the pitch and yaw engines are located approximately 73 ft fore and aft of the vehicle cg as shown in Figure 3-6. The roll engines are located approximately 7.5 ft from the vehicle cg. The high aerodynamic heating loads during entry require that the thrusters that point downward and those that point forward be covered during entry.

Table 3-2. Space Station Logistics Attitude Control

Requirement	Limit Cycle (deg)	ΔV (fps)	Duration	Number of Thrusters	Elapsed Time	% On Time	Interval Between Pulses (sec)	ΣΔV
Sensor Orientation Between 40 & 100 n.mi.	±45	0.33	40 min	1 Roll		0.026	960	
				1 Pitch		0.011	2250	
				1 Yaw		0.011	2250	
					45 min			0.33
Orientation Prior to Main Engine Burn	±5	1.5	5 min	1 Roll		0.234	107	
				1 Pitch		0.10	250	
				1 Yaw		0.10	250	
					50 min			1.83
Roll Control During Main Engine Burn at 100 n.mi.	±5	0.25	1 min	2 Roll		0.234	107	
					51 min			2.16
Sensor Orientation at 100 n.mi.	±45	8.7	18 hours (max.)	1 Roll		0.026	960	
				1 Pitch		0.011	2250	
				1 Yaw		0.011	2250	
					≈19 hr			10.86
Orientation Prior to Main Engine Burn	±5	1.5	5 min.	1 Roll		0.234	107	
				1 Pitch		0.1	250	
				1 Yaw		0.1	250	
					≈19.1 hr			12.36
Roll Control During Main Engine Burn at 100 n.mi.	±5	0.25	1 min.	2 Roll		0.234	107	
					19.1 hr			12.69
Sensor Orientation Between 100 & 260 n.mi.	±45	0.33	40 min.	1 Roll		0.026	960	
				1 Pitch		0.011	2250	
				1 Yaw		0.011	2250	
					19.7 hr			13.02
Orientation Prior to Main Engine Burn	±5	1.5	5 min.	1 Roll		0.234	107	
				1 Pitch		0.10	250	
				1 Yaw		0.10	250	
					19.8 hr			14.52
Roll Control During Main Engine Burn at 260 n.mi.	±5	0.25	1 min.	2 Roll		0.234	107	
					19.8 hr			14.77
Sensor Orientation at 260 n.mi.	±45	0.33	40 min.	1 Roll		0.026	960	
				1 Pitch		0.011	2250	
				1 Yaw		0.011	2250	
					20.7 hr			15.10
Orientation Prior to & During ACPS Engine Burn	±0.5	3.1	22 + sec	3 Roll		2.9	8.4	
				4 Pitch		1.28	19.5	
				2 Yaw		0.624	39	
					20.7 hr			18.20
ACPS Burn to Transfer From 260 to 270 n.mi.	-	16.9	10 sec	4 Longitudinal		100	0	
					20.7 hr			35.10
Sensor Orientation Between 260 & 270 n.mi.	±5	3.3	45 min.	1 Roll		0.234	107	
				1 Pitch		0.10	250	
				1 Yaw		0.10	250	
					21.5 hr			38.40
Orientation Prior to & During ACPS Engine Burn	±0.5	3.1	22+sec	3 Roll		2.99	8.4	
				4 Pitch		1.28	19.5	
				2 Yaw		0.642	39	
					21.5 hr			41.50

Table 3-2. Space Station Logistics Attitude Control, Contd

Requirement	Limit Cycle (deg)	ΔV (fps)	Duration	Number of Thrusters	Elapsed Time	% On Time	Interval Between Pulses (sec)	Σ ΔV
ACPS Burn to Circularize at 270 n.mi.	-	16.9	10 sec	4 Longitudinal	21.5 hr	100	0	58.40
Rendezvous	±5	1.1	15 min.	1 Roll		0.234	107	
				1 Pitch		0.10	250	
				1 Yaw		0.10	250	
					21.75 hr			59.50
Translation	-	7.2		4 Pitch				
				4 Yaw				
								66.70
Docking	±0.5	1.7	20 sec	3 Roll		2.99	8.4	
				4 Pitch		1.28	19.5	
				2 Yaw		0.642	39	
					21.75 hr			68.40
Space Station Staytime			6 days		7 days			68.40
Separation	±0.5	1.7	20 sec	3 Roll		2.99	8.4	
				4 Pitch		1.28	19.5	
				2 Yaw		0.642	39	
					7 days			70.10
Translation	-	8.3	10 sec	4 Pitch				
				4 Yaw				
					7 days			78.40
Sensor Orientation Prior to Entry	±45	11.6	24 hr	1 Roll		0.026	960	
				1 Pitch		0.011	2250	
				1 Yaw		0.011	2250	
					8 days			90.0
Orientation Prior to De-orbit Burn	±5	1.5	5 min.	1 Roll				
				1 Pitch				
				1 Yaw				
					8 days			91.5
Roll Control During Main Engine Burn @ 270 n.mi.	±5	0.25	1 min.	2 Roll		0.234	107	
					8 days			91.75
Vehicle Orientation Prior to Entry	±5	9.75	≈30 min.	1 Roll		0.234	107	
				1 Pitch		0.10	250	
				1 Yaw		0.10	250	
					8 days			101.5
Entry Phase I	±2	23.1	250 sec	4 Roll		1.87	13.3	
				8 Pitch		1.6	15.6	
				8 Yaw		1.6	15.6	
					8 days			124.6
Entry Phase II	±2	260	500 sec	3 Roll		1.4	17.8	
				6 Pitch		1.2	20.8	
				6 Yaw		1.2	20.8	
					8 days			150.6
Entry Phase III	±2	11.5	750 sec	1 Roll		0.47	53.3	
				4 Pitch		0.80	31.2	
				3 Yaw		0.60	41.6	
					8 days			162.10

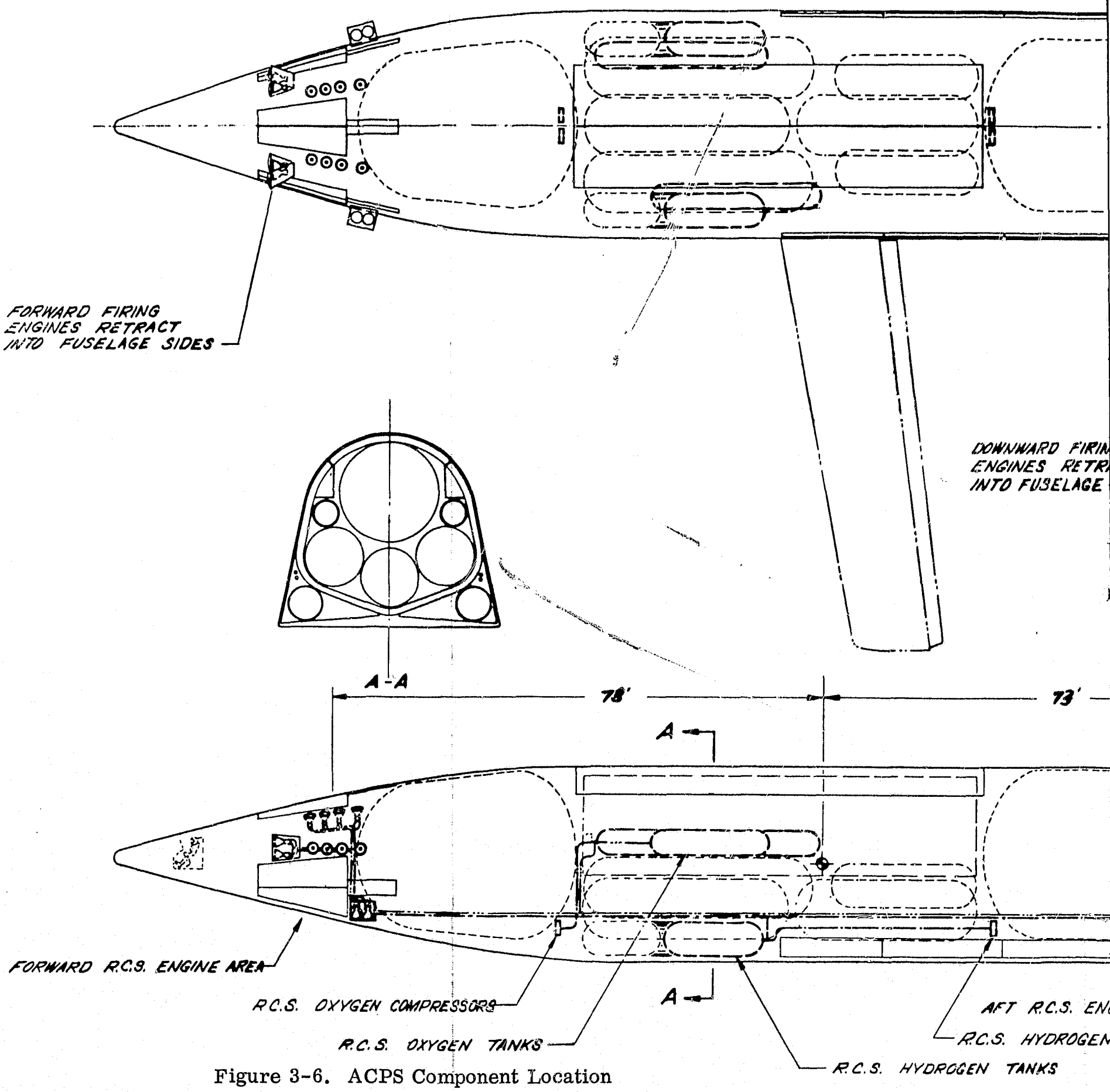
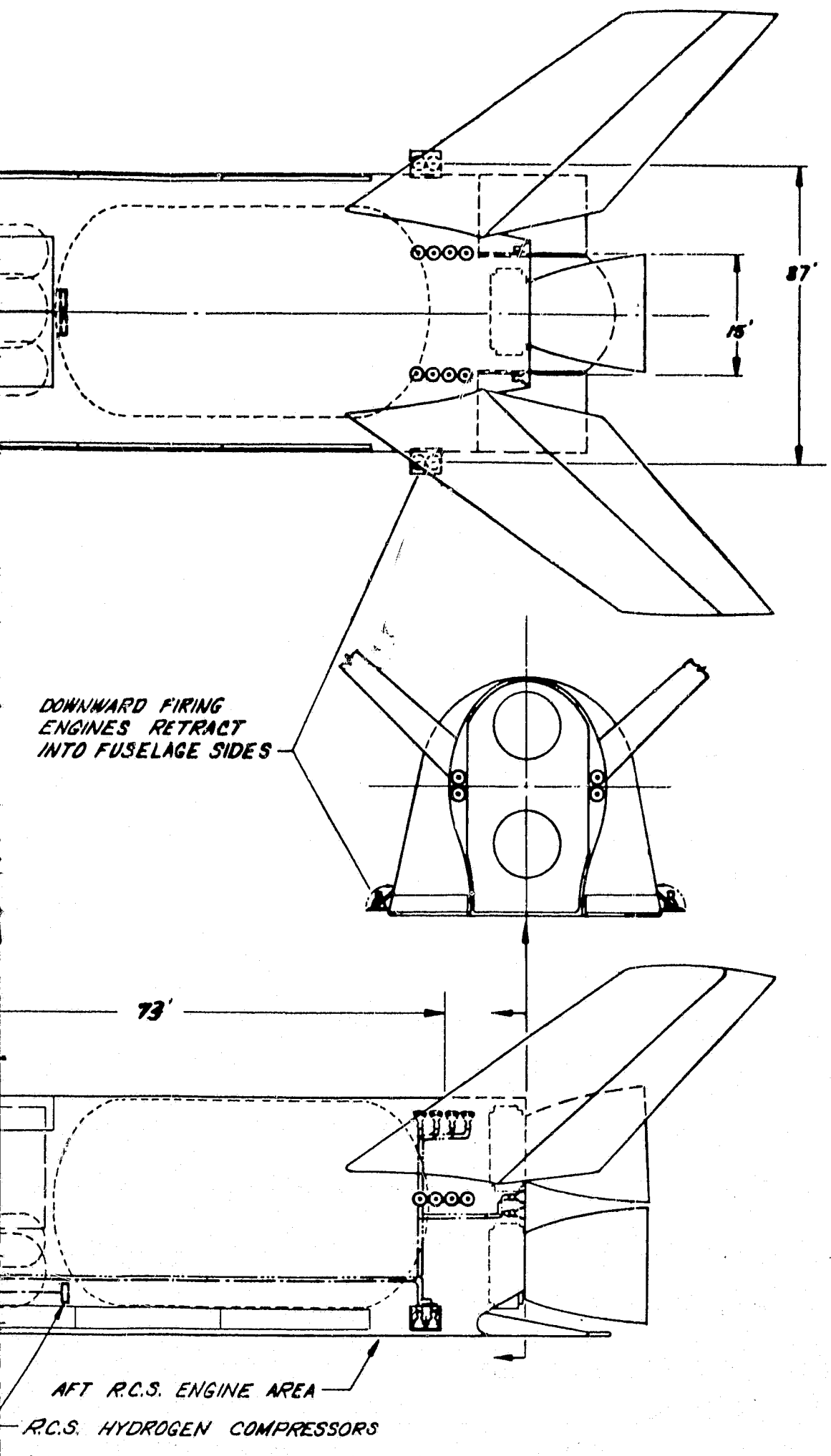


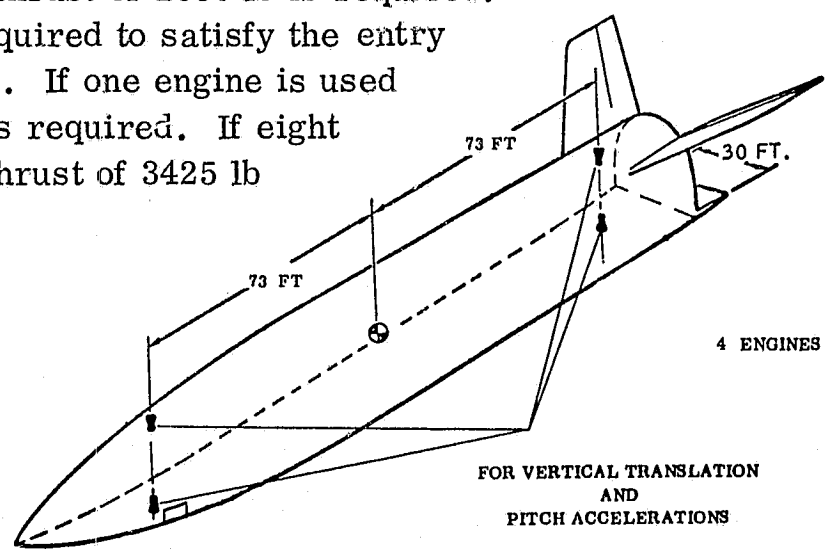
Figure 3-6. ACPS Component Location

FOLDOUT FRAME



FOLDOUT FRAME 2

3.1.5 ENGINE SIZE REQUIREMENTS. Table 3-3 shows the minimum acceptable thrust requirements in order to provide the orbital maneuver requirements of 0.03 g for a vehicle weight of approximately 408,000 pounds, which is a typical orbiter weight at initial injection at 100 n.mi. The inertia of the vehicle used in establishing these thrust requirements was 45.5×10^6 slug-ft^2 for pitch and yaw and 2.02×10^6 slug-ft^2 for roll. If one thruster is used for linear translations, a thrust of 12,400 lb is required to provide 0.03 g. If eight are used, a thrust of 1550 lb is required. Also shown are the minimum thrusts required to satisfy the entry maneuver requirements of 2.5 deg/sec^2. If one engine is used for pitch or yaw, a thrust of 27,400 lb is required. If eight engines are used to yaw the vehicle, a thrust of 3425 lb per thruster is required.

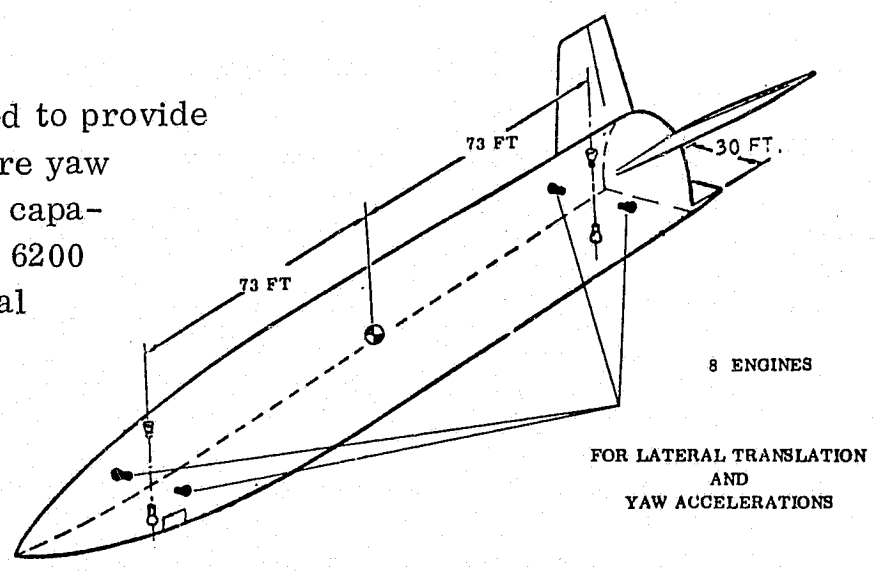


3.1.6 ENGINE ARRANGEMENTS. A minimum of 4 engines are needed to provide vertical translation capability and a pure pitch rate. If the engines have 6200 lb of thrust, the 0.03 g translational capability will be satisfied.

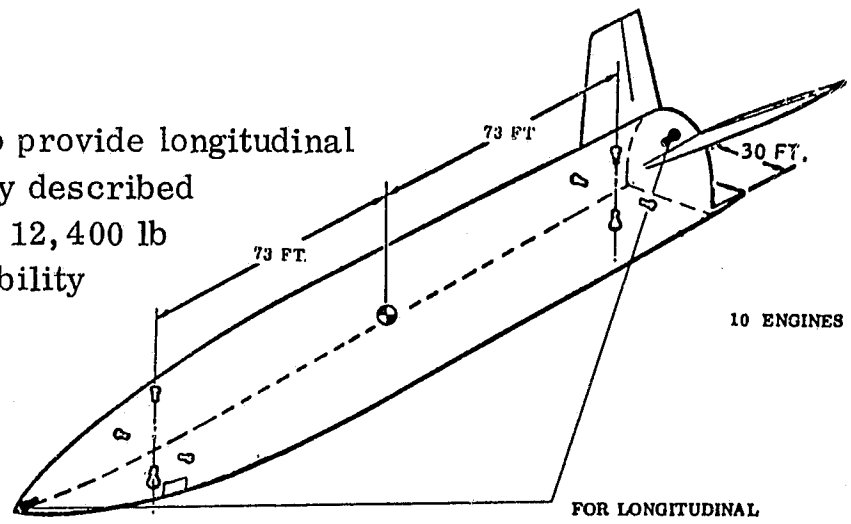
Table 3-3. Minimum Thrust Requirements

Number of Engines	Thrust Required (lb) Orbital Phase (0.03 g)	Entry Phase (2.5 deg/sec^2) Pitch, Yaw	Roll
1	12,400	27,400	11,700
2	6,200	13,700	5,850
4	3,100	6,850	2,925
8	1,550	3,425	1,463

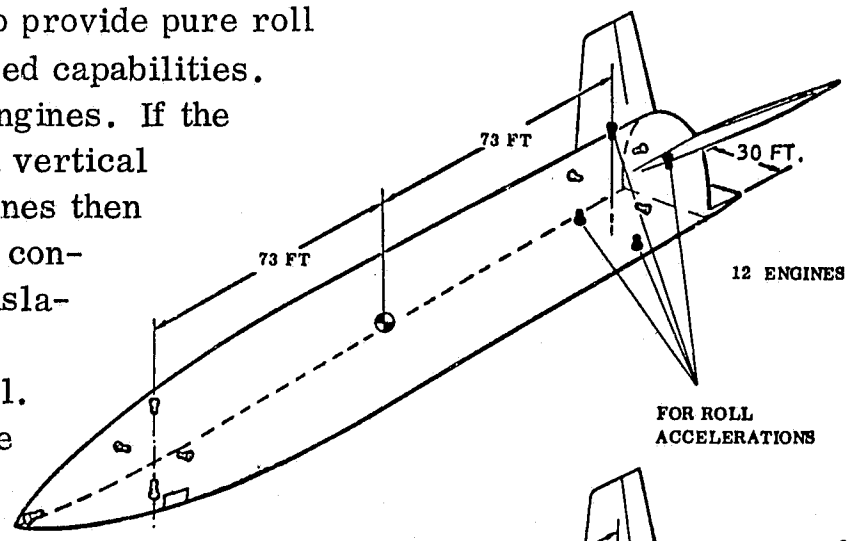
A minimum of 8 engines are needed to provide lateral translation capability, a pure yaw rate, and the previously described capabilities. If the added engines have 6200 lb of thrust, the 0.03 g translational capability will be satisfied.



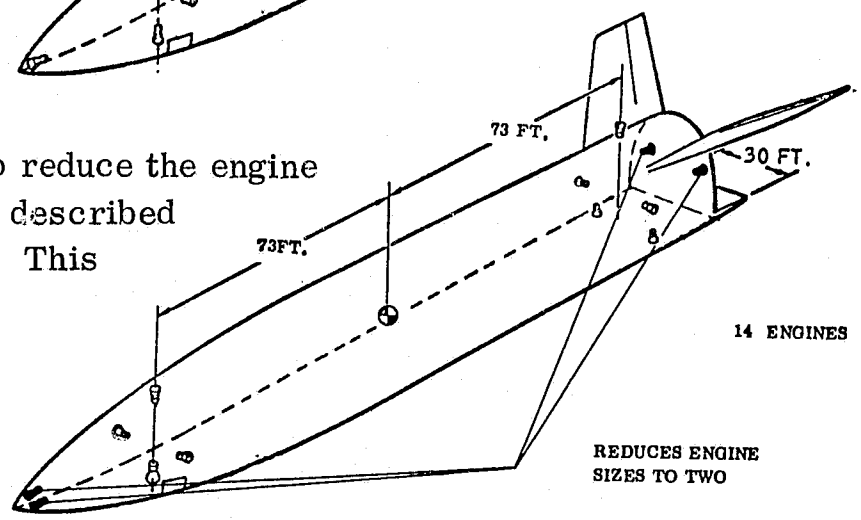
A minimum of 10 engines are needed to provide longitudinal translation capability and the previously described capabilities. If the added engines have 12,400 lb of thrust, the 0.03 g translational capability will be satisfied.



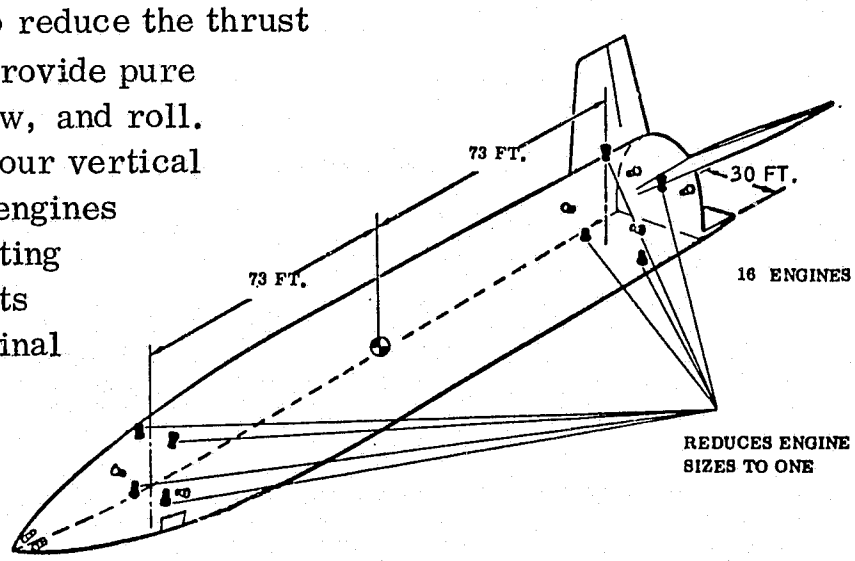
A minimum of 12 engines are needed to provide pure roll acceleration and the previously described capabilities. Four engines replace two aft vertical engines. If the engines are to be used with the forward vertical 6200-lb engines, each of the added engines then is required to be 3100-lb thrust. This configuration thus satisfies the 0.03 g translational capability in all three directions and pure couples in pitch, yaw, and roll. However, it requires engines with three different thrust levels: 12,000 lb, 6200 lb, and 3100 lb.



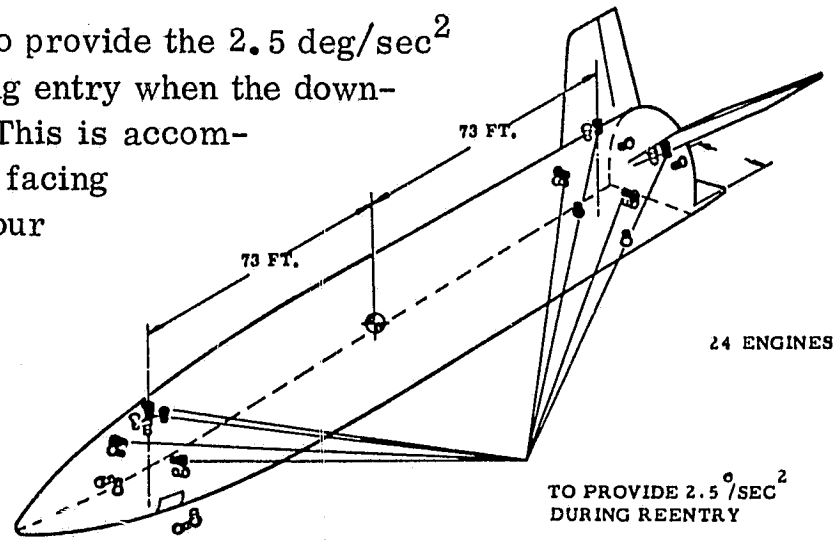
A minimum of 14 engines are needed to reduce the engine sizes to two and provide the previously described capabilities; i.e., 6200 lb and 3100 lb. This is accomplished by replacing the two engines of 12,400-lb thrust which face fore and aft with four engines of 6200-lb thrust.



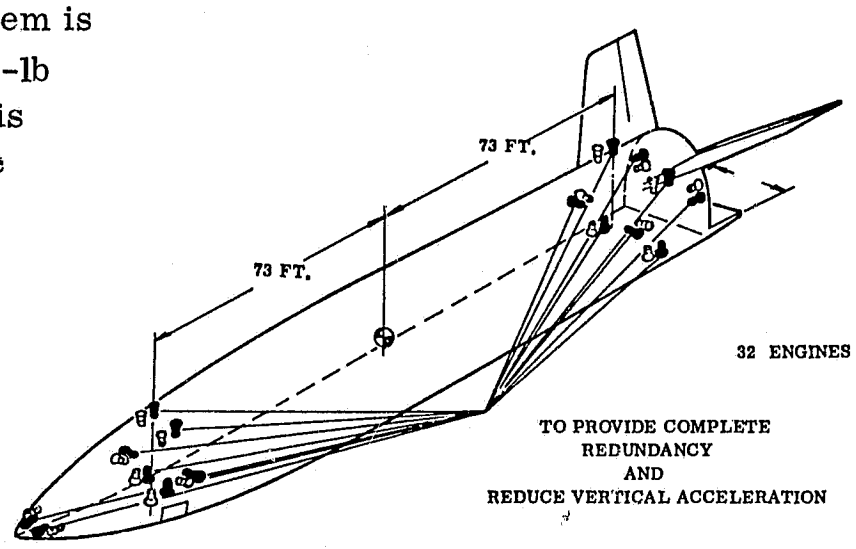
A minimum of 16 engines are needed to reduce the thrust size to one (i.e., 6200-lb thrust) and provide pure translation or pure couples in pitch, yaw, and roll. This is accomplished by replacing the four vertical aft engines of 3100-lb thrust with four engines of 6200-lb thrust and adding and relocating two vertical forward engines with thrusts of 6200 lb. The horizontal and longitudinal accelerations remain at 0.03 g but the vertical acceleration is increased to 0.06 g.



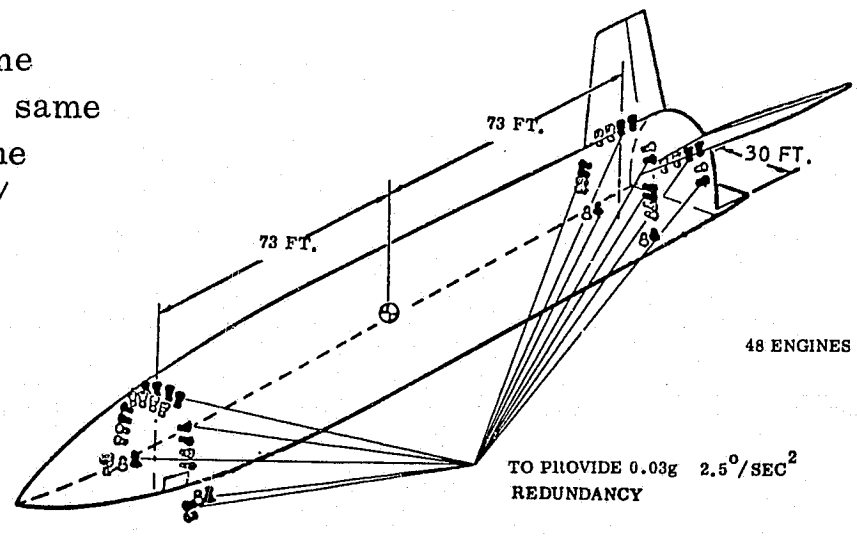
A minimum of 24 engines are needed to provide the 2.5 deg/sec^2 pitch, yaw, and roll acceleration during entry when the downward facing engines are inoperative. This is accomplished by adding four vertical engines facing upward (two forward and two aft) and four yaw engines (two forward and two aft). It also requires that the engine thrust level be increased to 6850 lb. This subsystem has 24 engines. The minimum vertical acceleration remains at 0.06 g, which exceeds the translational upper limit.



A derivative of the 16-engine subsystem is characterized by replacing each 6200-lb engine with two 3100-lb engines. This subsystem is capable of providing the required translational maneuver capability of 0.03 g, but provides less than the required rotational maneuver capabilities in pitch and yaw for entry (only 1.2 deg/sec^2). It also provides pure translation and rotation with engines inoperative, but at reduced acceleration.



A 48-engine derivative of the 24-engine subsystem is capable of providing the same maneuver capabilities as the 24-engine subsystem, i.e., 0.03 g and 2.5 deg/sec^2. The subsystem is characterized by replacing each of the 6850-lb engines with two 3425-lb engines. Vehicle control is thus provided with engines inoperative, but at reduced accelerations.



The above discussion shows that the number of thrusters can vary from 12 to 48, depending on the capability desired. Additional studies of missions, ACPS duty cycle, limit cycle constraints, reliability, cost, minimum impulse bit, valve power requirements, packaging, and weight are required in order to arrive at the final selection.

For the remainder of the discussion, 48 engines are considered at a thrust level of 3425 lb since this system provides all the identified requirements:

a. 0.03 g translational acceleration.

b. Pure translation and rotation for docking.

c. 2.5 deg/sec^2 angular acceleration.

d. Redundancy with no crosscoupling during docking but at reduced acceleration rates.

e. Forward and downward facing engines covered during entry.

3.1.7 CHAMBER PRESSURE/EXPANSION RATIO CONSTRAINTS. Figure 3-7 shows a typical installation of two 3425-lb-thrust engines with expansion ratios of 50* and chamber pressures of 500 psia (left) and 300 psia (right) indicating that chamber pressures of less than 300 psia would be difficult to accommodate. For stable engine operation, inlet pressure in the order of 100 psia greater than the chamber pressure is required, dictating supply pressures of 400 psi or greater.

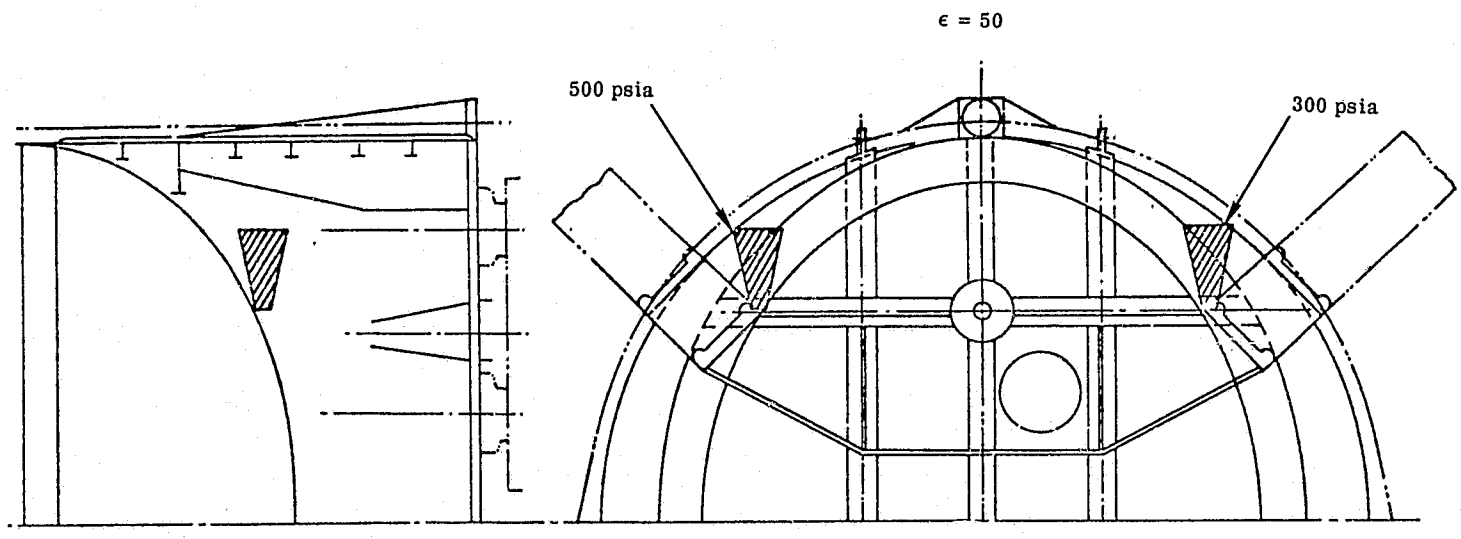


Figure 3-7. Typical Engine Installation

3.2 OXYGEN-HYDROGEN ENGINE PERFORMANCE

Figure 3-8 shows the theoretical vacuum specific impulse for an oxygen-hydrogen engine as a function of mixture ratio and propellant initial inlet temperatures. For a mixture ratio of 4, propellant inlet temperatures between 200° F and 500° R, and a chamber pressure on the order of 500 psia, theoretical vacuum specific impulse values of approximately 470 seconds are indicated. Data provided by Rocketdyne indicates that approximately 94% of the theoretical I_{sp} may be delivered (i.e., 440 seconds) for steady state operation.

*Selected based on using gas accumulators supply systems.

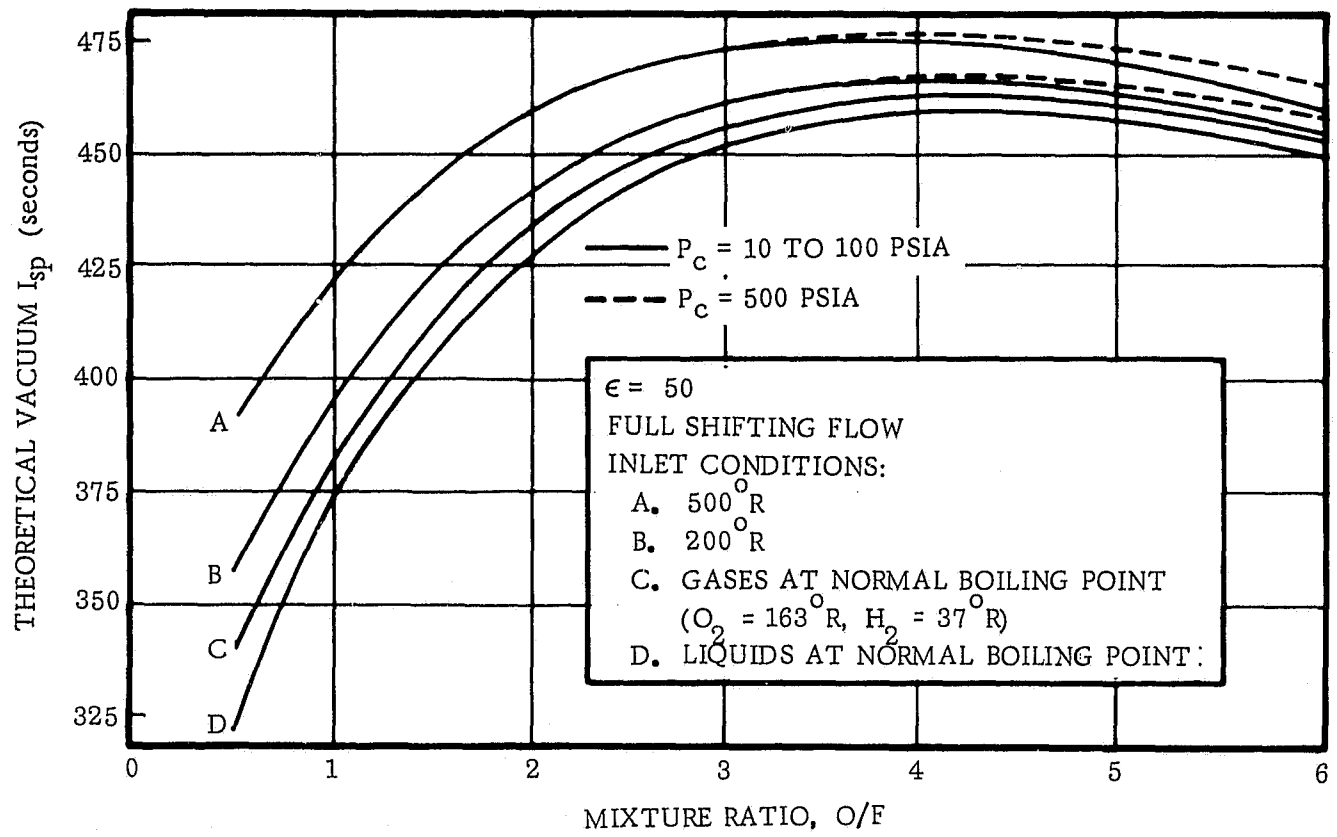


Figure 3-8. Theoretical Shifting Vacuum Specific Impulse as a Function of Mixture Ratio for Four Inlet Conditions

Evaluation of vehicle control requirements have established (see preceding section) that (1) the minimum pulse duration is in the order of 250 milliseconds and (2) the majority of the propellant will be expended in pulse durations on the order of 250 milliseconds. Data provided by Rocketdyne for storable propellants indicates that for pulse duration in the order of 250 milliseconds to 1 second the delivered specific impulse in the pulsing mode is approximately 95% of the steady state specific impulse. Applying this 0.95 modifier, a value of 420 seconds was estimated. (Limited data on O_2-H_2 engines indicate the gaseous O_2-H_2 propellant may have better pulse performance than the storable propellants.

Several ignition concepts are being considered for the O_2-H_2 ACPS; e.g., catalyst, spark, additive. Current ignition technology for a catalyst requires that the average propellant inlet temperature passing through the catalyst be at 200° R or greater to achieve prompt, repeatable ignition.

3.3 ENGINE-CHARGED, CONSTANT THRUST SUBSYSTEM

With an average specific impulse of approximately 420 seconds, average vehicle weight of approximately 370,000 lb, and ΔV requirements of 15 and 50 fps, propellant quantities of 400 lb and 1340 lb are required for drag makeup and entry for the low earth orbit mission (Figure 3-1).

With the thrust value, minimum pressure level, propellant quantity, and minimum temperature values established, a minimum subsystem was identified to satisfy the requirements. It is shown in Figure 3-9. Gaseous oxygen and hydrogen are bled from the engine during boost phase, charging storage bottles to 1000 psia. The final temperatures of the charged gases are approximately 500° R for oxygen and 170° R for hydrogen.

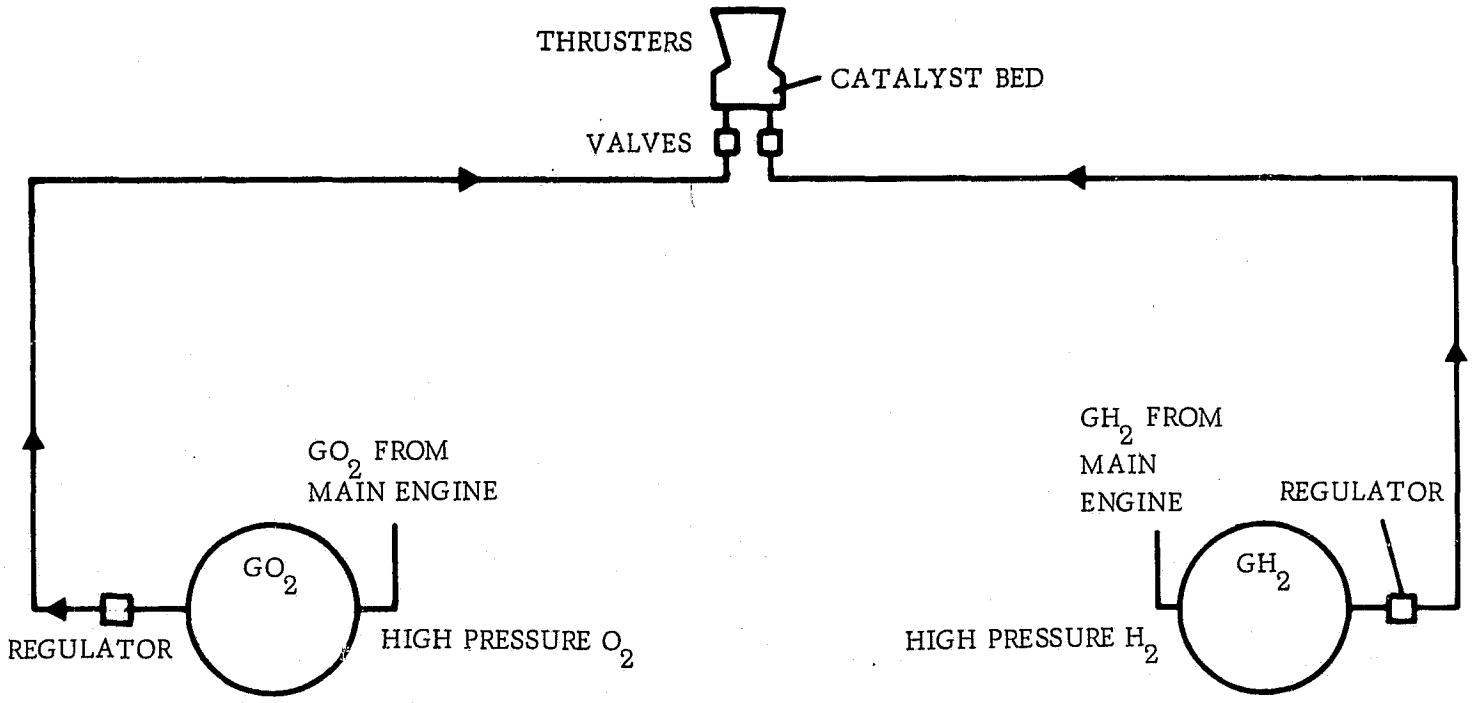


Figure 3-9. Propellant Supply Subsystem

The temperature of the hydrogen is maintained at approximately 170° R by connecting the gaseous hydrogen tank of the ACPS with the liquid oxygen tank of the orbit maneuvering propulsion subsystem with a heat pipe. Both the gaseous hydrogen and liquid oxygen tanks are mounted and insulated to minimize heat flow from the surroundings. The gaseous oxygen is at a temperature of approximately 500° R in equilibrium with the vehicle structure and accordingly does not require insulation.

As the mission progresses, the propellant needed for drag makeup is provided to the engines. During the de-orbit burn using the main engine, the tanks are recharged using bleed gas.

In order to have 1340 lb of propellants available at a pressure of 400 psia to maintain a thrust level of 3425 lb during the entire entry, 2240 lb must be charged in the tanks*. At a mixture ratio of 4 to 1, the hydrogen gas loaded is 450 lb and the oxygen is 1790 lb. The volume of the hydrogen container is 340 ft^3 and the volume of the oxygen tank is 310 ft^3. Space available for installation dictates the use of cylindrical tanks. The weight of two cylindrical steel hydrogen tanks is 2650 lb. The weight of two cylindrical steel oxygen tanks is 2425 lb.

*As seen in Figure 3-9, this may not be required. This does, however, establish a boundary case.

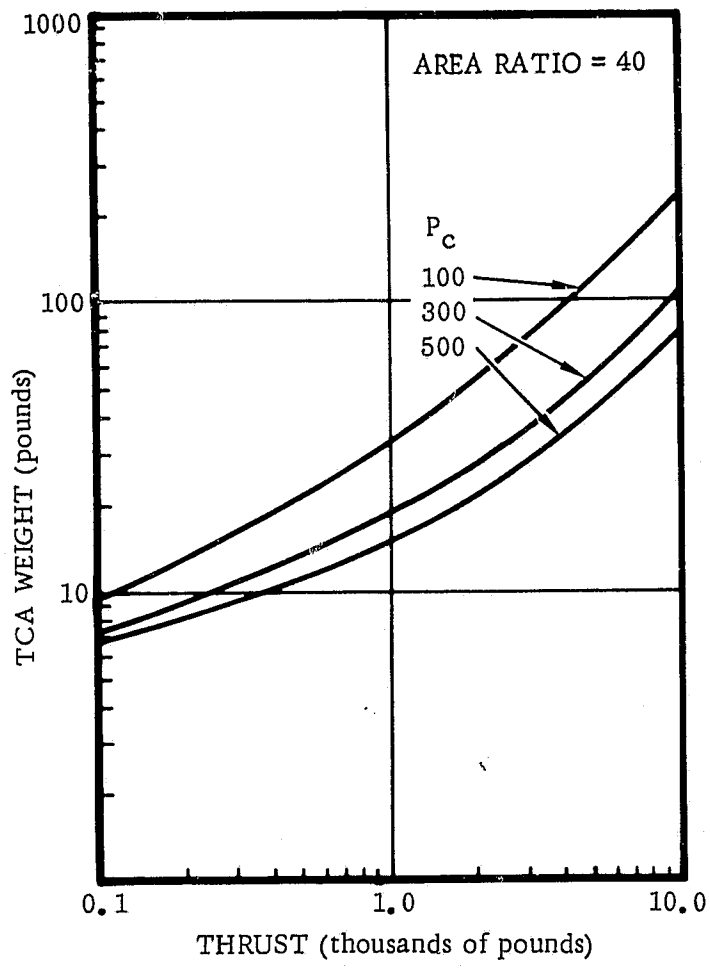


Figure 3-10. Engine Weight as a Function of Thrust

Figure 3-9 shows the general line arrangement which supplies the gas from the accumulator to the engines. For two hydrogen lines and two oxygen lines, a total line of 480 ft is required. Based on maintaining a Mach number of 0.3 in the lines to minimize pressure drop, steel lines approximately two inches in radius are required with wall gages of approximately 0.020 inch. The total line weight is approximately 600 lb including allowance for regulators, valves, connections, and mounting.

Data provided by Aerojet General (Figure 3-10), indicates that engines with thrusts of 3425 lb and 300 psia chamber pressure will weigh approximately 43 lb. For 48 engines, they will have a total weight of 2100 lb. Table 3-4 summarizes the weight of this oxygen-hydrogen ACPS.

Table 3-4. Weight Summary for Pump-Charged H_2-O_2 ACPS for Low Earth Orbit Mission (Constant Thrust)

	Weight (lb)
Required Propellants	
ΔV 50 fps	1,340
ΔV 15 fps	400
Residuals	900
	2,640
Tankage	
Hydrogen	2,650
Oxygen	2,425
	5,075
Lines, Regulators	600
Engines	2,100
Total	10,415

3.4 ENGINE CHARGED, REDUCING THRUST DURING ENTRY

If the engines and tanks are allowed to operate at reduced inlet pressures and reduced thrust levels during the later phase of entry, as suggested by Figure 3-5, the propellant residual and tank weight can be reduced resulting in a subsystem weight in the order of 7720 lb. The weight breakdown of this subsystem is shown in Table 3-5.

Table 3-5. Weight Summary for Engine-Charged H_2-O_2 ACPS for Low Earth Orbit Mission (Reducing Thrust)

	Weight (lb)
Required Propellants	
ΔV 50 fps	1340
ΔV 15 fps	400
Residual	100
	1840
Tankage	
Hydrogen	1620
Oxygen	1560
	3180
Lines	600
Engines	2100
Total	7720

3.5 COMPRESSOR CHARGED SUBSYSTEMS

For missions with greater on-orbit maneuver capability, improved subsystems were identified. These are shown in Figures 3-11, 3-12, and 3-13. These subsystems were designed to permit use of residual and boiloff propellants which are available in significant quantities, as shown in Table 3-6. For each pound of propellant compressed for use by the subsystem shown in Figure 3-11, approximately 0.1 lb of propellant is used in the fuel cell to generate compressor power. The propellant used in the fuel cell results in water which may be considered a useful payload in some missions. For each pound of propellant compressed by the subsystems shown in Figures 3-12 and 3-13, approximately 0.2 lb and 0.3 lb of propellant, respectively, are used but no usable water is produced. The three subsystems provide the potential propellant ΔV capability and water as shown in Table 3-7.

The weight of the fuel cells, electric motors, and compressors has been estimated to be in the order of 300 lb, with a significant portion of the weight being required for the fuel cells. Weights of the reciprocating engine and compressor or turbine compressor are currently being evaluated.

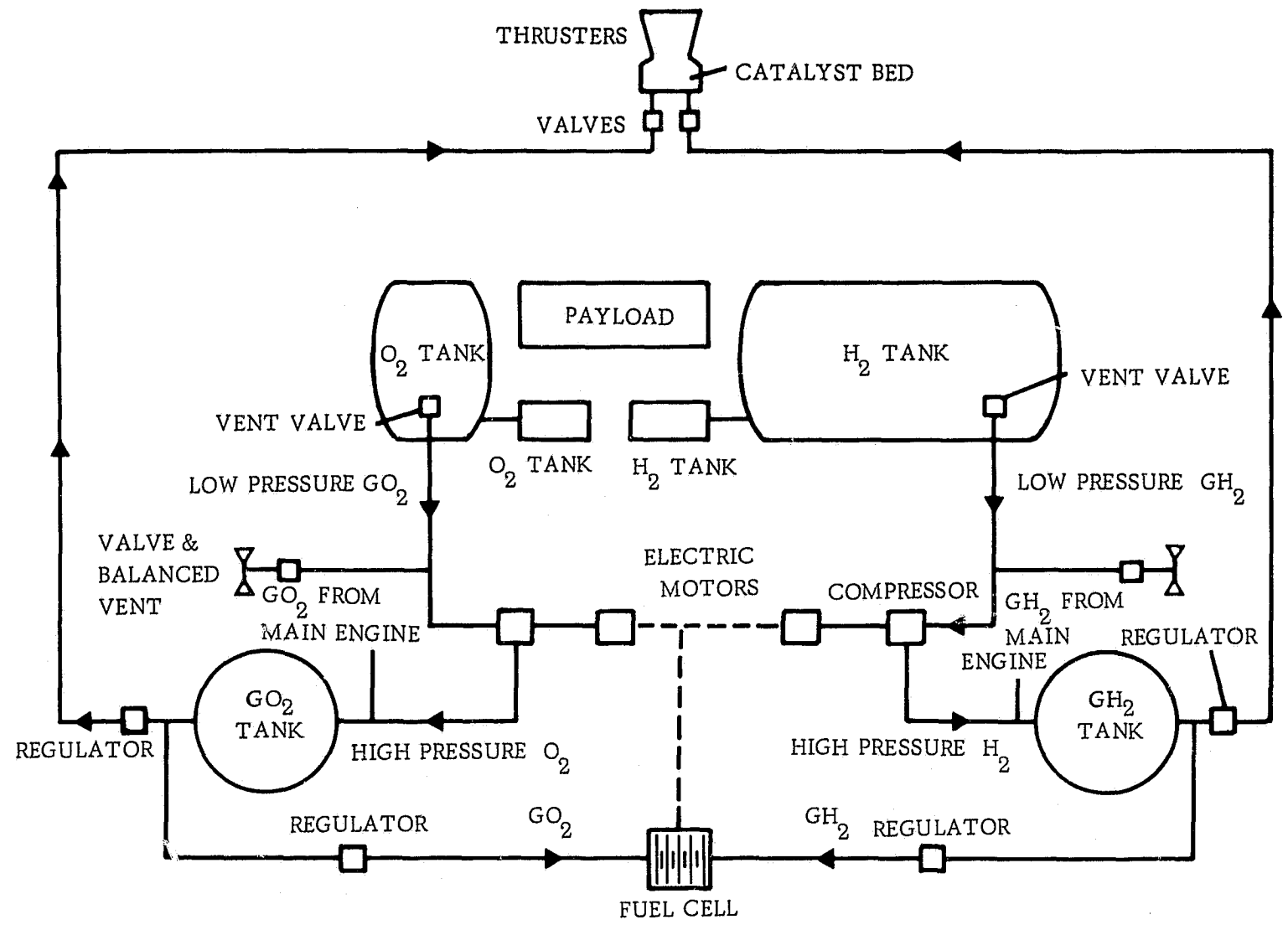


Figure 3-11. Compressor-Charged ACPS

Table 3-6. Orbiter Element Residuals and Boiloff

Oxygen Residuals	
Liquid in Main Tank	1,040
Gas in Main Tank	2,910
Gas in Maneuver Tank	291
Liquid in Line	4,575
	8,816
Hydrogen Residuals	
Liquid in Main Tank	69
Gas in Main Tank	527
Gas in Maneuver Tank	415
Liquid in Line	343
Hydrogen Boiloff	3,737
	5,098
Total	14,012

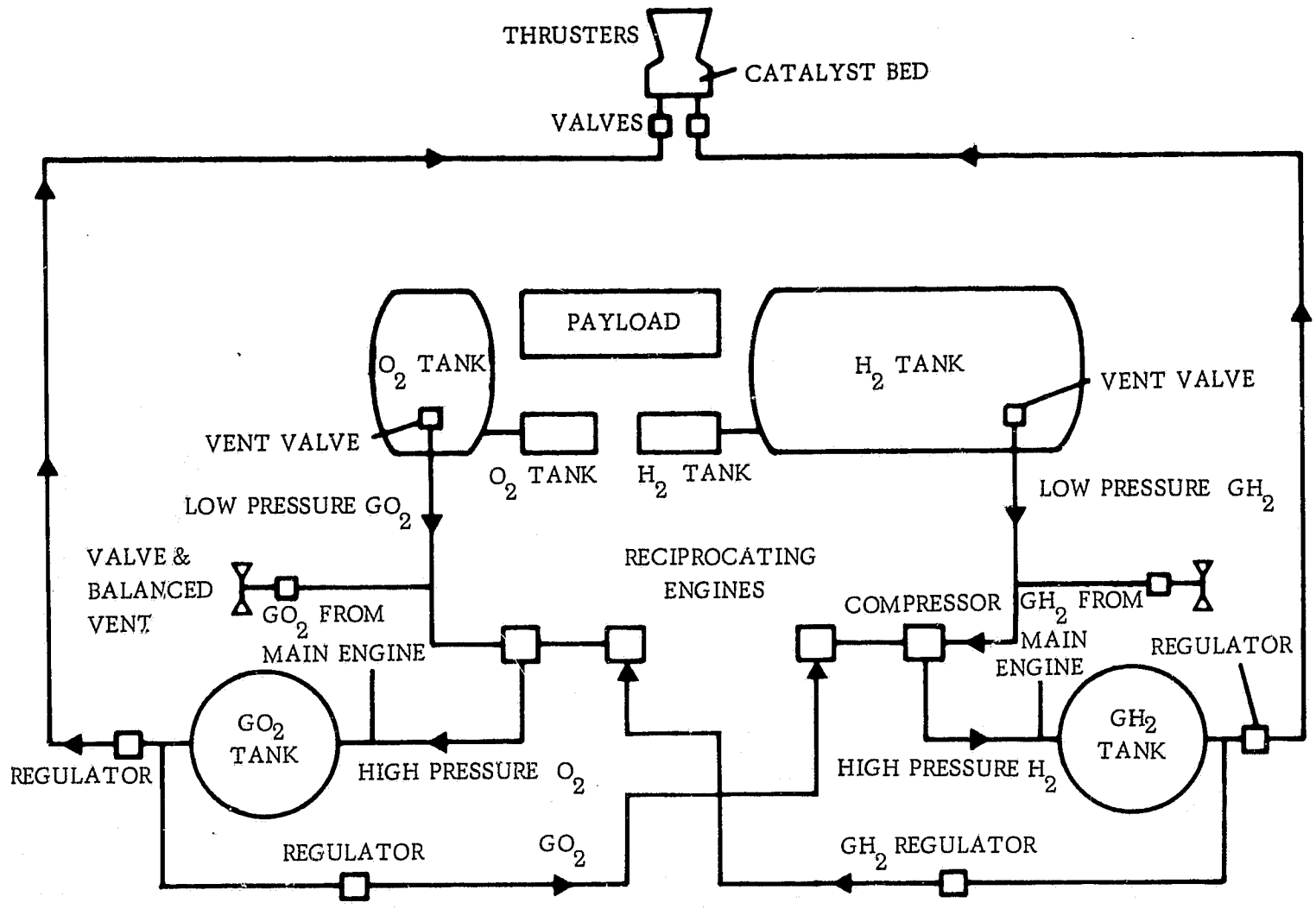


Figure 3-12. Compressor-Charged ACPS

Table 3-7. Potential ΔV and Water Capabilities

	ACPS Propellant*	ACPS ΔV (fps)	Water (lb)
Fuel Cell Subsystem	12,700	475	1300
Reciprocating Engine	11,600	430	0
Combustor Turbine	10,800	400	0

*May be off-loaded for space station engines if not required by space shuttle.

In summary, an oxygen-hydrogen ACPS using residuals and boiloff has the potential of providing ΔV of 540 fps if all residuals are consumed (65 fps on original engine charge and 475 on compressor charge) for a subsystem weight of 8020 lb (7720 + 300).

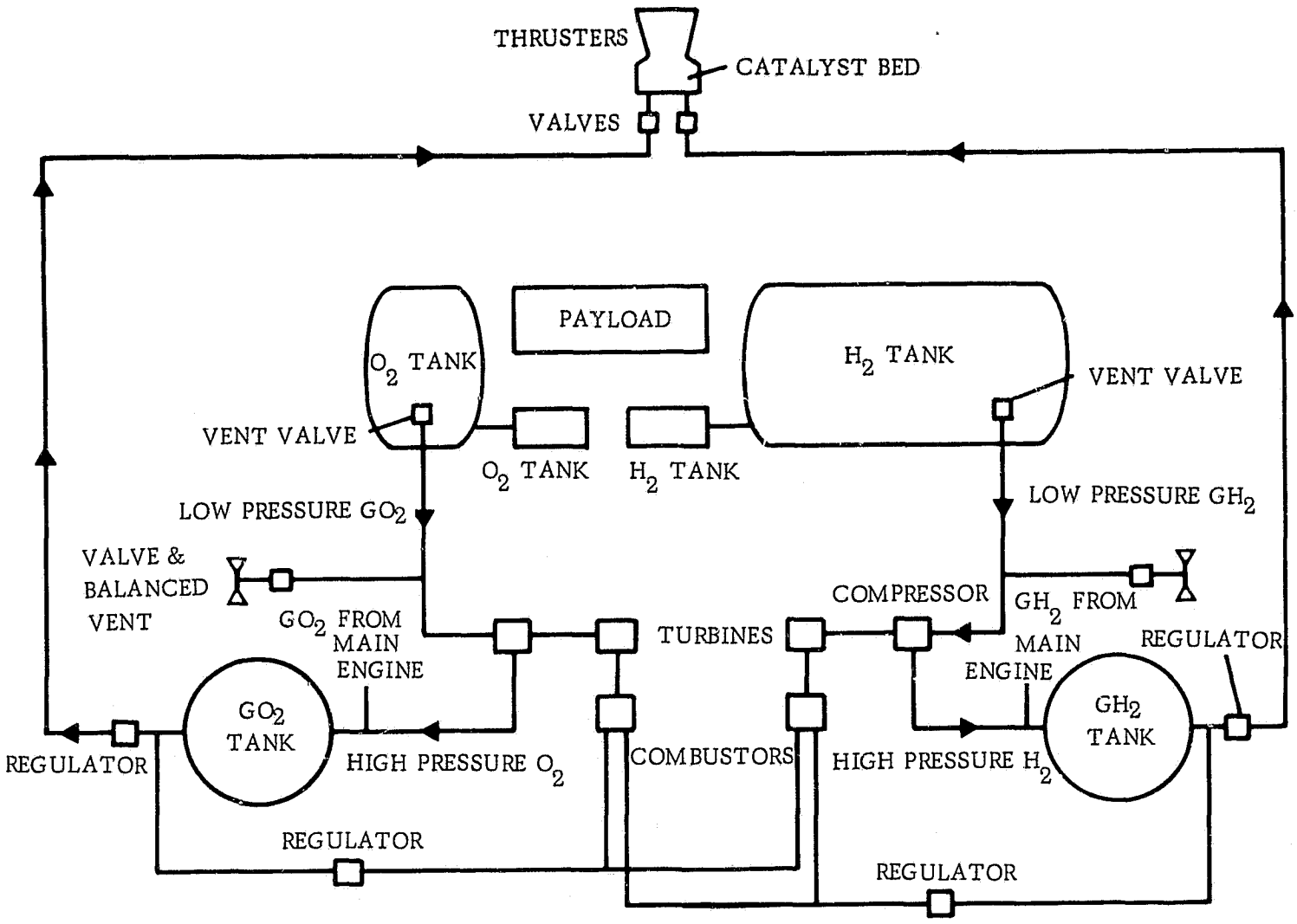


Figure 3-13. Compressor-Charged Subsystem

3.6 LIQUID H_2-O_2 SUBSYSTEMS

An ACPS which utilizes liquid propellants is being considered for the space shuttle vehicle. At the present time, the subsystems which require a liquid supply appear less desirable than the gaseous supply subsystem for the following reasons:

a. The liquid supply subsystem cannot utilize gaseous residuals and boiloff.

b. There is the problem of assuring pure liquid in the zero-g environment, considering engine heat soak-back, engine cooling, and burnout.

c. Response and ignition requirements appear to dictate the use of relatively warm gaseous propellants at the engine.

In future studies, these subsystems should be investigated further.

3.7 OXYGEN-HYDROGEN/EARTH-STORABLE PROPELLANT ACPS COMPARISON

During the initial portion of the study, storable-propellant attitude control propulsion subsystems were utilized because they were state-of-the-art. During the initial study it became apparent that the storable-propellant ACPS would have a number of problems:

a. Limited flexibility.

b. High recurring cost.

c. Potentially significant corrosion problems resulting from the $N_2O_4 + H_2O \rightarrow HNO_3$ nitric acid reaction.

d. Contamination.

e. Thermal control (i.e., freezing).

f. Technology problems, e.g., reuseable light weight positive expulsion devices for large tanks, reuseable nondetonatable thrust chamber.

g. Servicing of toxic systems.

h. High weight.

Accordingly, the oxygen-hydrogen ACPS was considered.

Table 3-8 is a comparison between the oxygen-hydrogen subsystem and the earth-storable subsystem for a ΔV of 200 fps. Subsystem weights for other ΔV's are shown in Figure 3-14.

Table 3-8. Oxygen-Hydrogen/Storable Propellant ACPS Weight Comparison $\Delta V = 200$ ft/sec at 300 psia for Both Systems

	O_2-H_2 (lb)	Storable (N_2O_4/MMH) (lb)
Pressurant Subsystem		
Pressure Tank	0	750
Regulators, Valves, Etc.	0	96
Pressurant	0	71
Propellant Subsystem		
Fuel Tank	1,620	200
Oxidizer Tank	1,560	184
Valves and Filter	50	38
Lines	550	231
Propellants		
Fuel	370	3,600
Oxidizer	1,470	5,399
Thrusters	2,100	1,872
Inlet, Compressor, and Power Supply	300	
Total	8,020	12,441

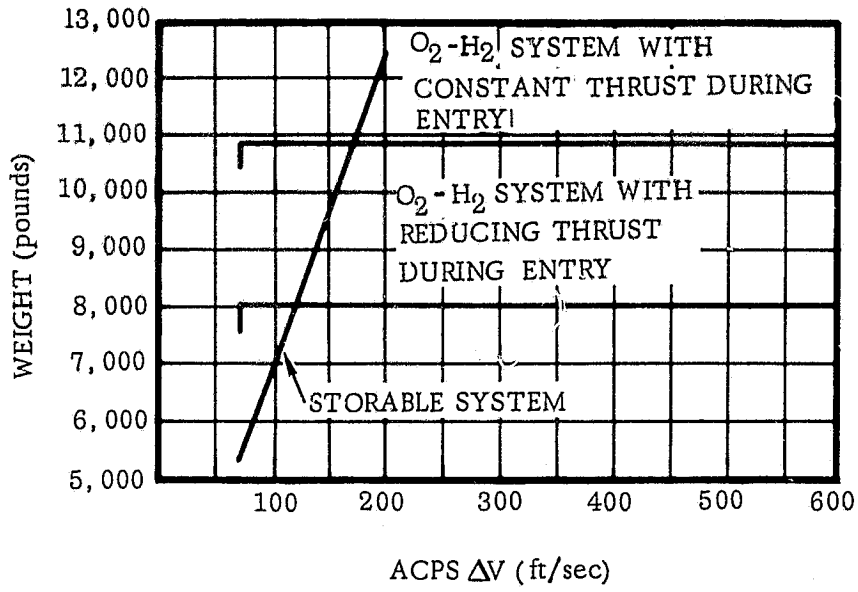


Figure 3-14. ACPS Weight Comparison for a Range of ΔV

The oxygen-hydrogen subsystem has a slight weight penalty over the earth-storable subsystem for ΔV's in the order of 100 fps. For ΔV's in the order of 200 fps or greater, the O_2-H_2 subsystem has a weight advantage.

Other factors favoring an oxygen-hydrogen ACPS are low recurring cost, flexibility, noncorrosiveness, noncarboneous exhausts, non-toxicity, elimination of bladders, and elimination of electric heaters and their power subsystem requirements to prevent propellant freezing.

Considering the above factors, the oxygen-hydrogen ACPS appears to be a logical choice for a space shuttle/space transportation system.

3.8 SUGGESTIONS AND RECOMMENDATIONS

Additional studies of the subsystem are suggested. Areas which appear worthy of further investigation include:

a. Effect of relative compressor and accumulator sizes.

b. Chamber pressure optimization.

c. Better definition of thrusters, including transients; effects of inlet condition variations on thrust, mixture ratio, I_{sp}, valve powers, minimum impulse bits, pulsed performance, cooling, ignition rise time, shutdown impulse variations and types of valving.

d. Compressor power supply sources.

e. Mission duty cycle definition.

f. Regulation and control devices.

g. Utilization of an auxiliary power supply for compressor drive.

h. Common use of hydrogen compressors for airbreathing engines and ACPS.

i. Residual gas availability and temperatures.

j. Higher main tank pressures for direct thruster supply.

k. Effects of main engine settling and chilldown requirements on ACPS propellant requirements.

l. Effects of residual gas temperature on compressor cost, weight, volume, propellant consumption, liquid residuals, and tank weight.

m. Volume and weight effects of heat exchangers to regulate residual gas temperature.

3.9 REFERENCES

3-1 Rendezvous and Docking, General Dynamics Report GDC-DCE66-027.

3-2 Advances in Astronautical Sciences, Volume XVI, Part I, "Manned Control of Space Vehicle Docking," by J. E. Montgomery.

3-3 Communications Between O. C. Jean of NASA and G. Karel of General Dynamics Convair, June 1969.

3-4 Handling Requirements for Hyper-Velocity Aircraft, Cornell Specification TC-1332-F-1.

SECTION 4

AIRBREATHING FLYBACK AND LANDING PROPULSION

The space shuttle vehicle has airbreathing flyback and landing engines in both booster element and landing engines in the orbiter element. The booster element will fly back several hundred miles, while the orbiter element will enter the atmosphere in a manner requiring airbreathing engines for approach and landing only. This section:

a. Describes the required functions of the booster and orbiter airbreathing flyback and landing engines.

b. Gives the basis for determining the required thrust, fuel weights, engine weights, and engine dimensions for the different space shuttle configurations.

c. Describes several airbreathing engines that are or will be available during the next several years.

d. Shows installations of several candidate engines.

e. Discusses the operation and environmental problems of the airbreathing engines.

4.1 REQUIREMENTS

After first-stage burn, the booster element stages, reverses direction, enters, deploys the airbreathing engines, starts the engines, flies back, and lands at the launch site. The orbiter element continues into orbit, performs the scheduled mission and then enters the atmosphere, coasts back to the launch area, deploys the airbreathing engines, and starts them in time to make a powered approach and landing.

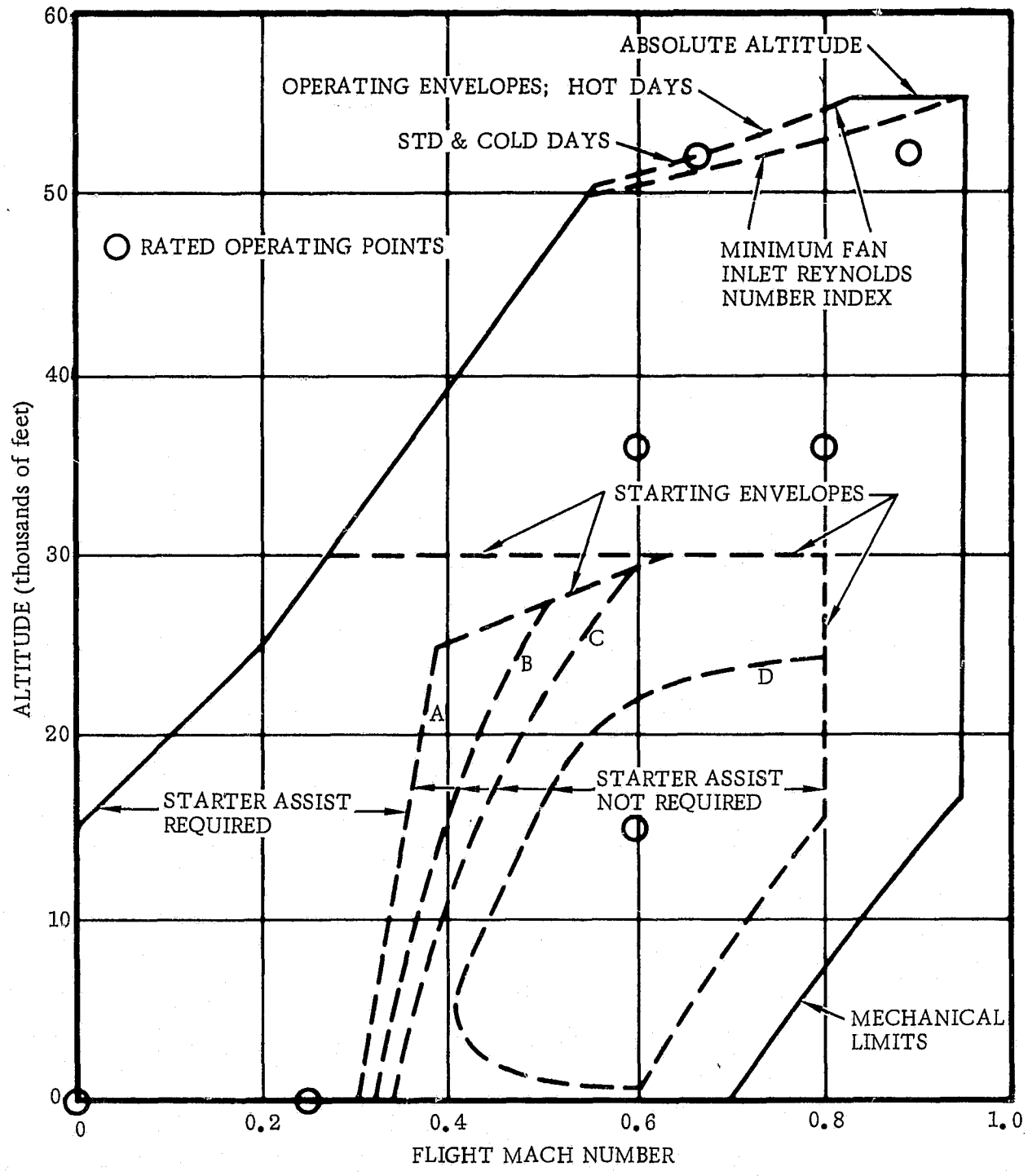
4.1.1 BOOSTER REQUIREMENTS. The requirements assumed for the booster element flyback engines include:

a. Windmill start.

b. Cruise back to launch site at 15,000 feet altitude.

c. Cruise back capability with engine out.

The booster engines will be deployed at an altitude of 25,000 feet or higher and will be windmill started during glide to the normal cruise altitude of 15,000 feet. Figure 4-1 shows the operating limits for a subsonic cruise turbofan engine. Windmill starting the engines at Mach 0.6 between 25,000 feet and 20,000 feet is compatible with these operating limits.



STARTER ASSIST NOT REQUIRED FOR:

A. GAS GENERATOR ROTOR WINDMILL SPEED HAS NOT BEEN ALLOWED TO DROP BELOW 600 rpm.

B. GAS GENERATOR ROTOR IS STOPPED AND SOAKED TO HOT-DAY AMBIENT TEMPERATURE BEFORE WINDMILL START ATTEMPT.

C. GAS GENERATOR ROTOR IS STOPPED AND SOAKED TO STANDARD DAY AMBIENT TEMPERATURE BEFORE WINDMILL START ATTEMPT.

D. GAS GENERATOR ROTOR IS STOPPED AND SOAKED TO COLD-DAY AMBIENT TEMPERATURE BEFORE WINDMILL START ATTEMPT.

Figure 4-1. Engine Operating Limits at 100 Percent Ram Air Efficiency

Under normal conditions, the booster element will cruise back at an altitude of about 15,000 feet. The velocity corresponding to the dynamic pressure at which an aircraft flies at a maximum lift-to-drag ratio (L/D) increases with altitude so that the flyback time decreases with altitude. This velocity is inversely proportional to the square root of the air density. As the velocity increases, however, the specific fuel consumption increases so that the net effect is a slight decrease in fuel weight as the altitude increases. The thrust delivered by an airbreathing engine decreases as velocity and altitude increase. The net result is an airbreathing propulsion subsystem weight that is a minimum but quite insensitive to altitude in the 10,000 to 15,000 feet altitude range. The altitude for minimum weight depends on the characteristics of the specific engine and vehicle.

A 15,000-ft cruise altitude, in addition to giving a near minimum weight system, provides the capability to flyback or perform a ferry mission over land if this becomes a requirement. It also gives engine-out capability for a configuration of three or more engines.

The booster is required to have the capability to cruise back in case of failure of one engine. If the booster has three engines sized for normal cruise at 15,000 feet, it will cruise at a maximum continuous emergency rating with one engine out at about 3,000 feet altitude. The greater the number of engines, the higher the cruise back altitude can be with engine out. The two-engine configuration would require the engines to be sized for the engine-out requirement. Since fuel consumption increases as altitude decreases the fuel weight provided must satisfy the engine-out condition.

4.1.2 ORBITER REQUIREMENTS. The minimum requirements assumed for the orbiter element landing engines include:

a. Windmill start.

b. Powered approach and landing capability at sea level.

c. Capability to withstand launch and space environment.

Additional requirements considered include:

a. Engine-out capability.

b. Capability to climb and go-around.

c. Commonality of engines on the orbiter and booster elements.

The orbiter engines will be deployed at an altitude of 15,000 feet or higher and windmill started during glide toward the landing site. Figure 4-1 indicates that a velocity of Mach 0.5 at 10,000 feet is within the windmill start capability limits of a subsonic cruise engine.

The engines will operate at idle to an altitude of about 1500 feet, where they will be brought to the thrust required for a powered approach and landing. The minimum capability required is for the engines to have only enough thrust at maximum power to fly level so that the glide path can be extended if necessary. Additional thrust may be required in order to have the capability to climb at a reasonable angle (4 to 5 degrees) for go-around and make a second approach and landing attempt. Reliability requirements may also impose the capability for level flight in case of the failure of one engine.

Some space shuttle configurations under consideration consist of two booster elements and an orbiter element that are as identical as possible considering the different mission requirements of the booster and orbiter elements. One of the ground rules of this configuration is common airbreathing engines on the booster and the orbiter. Sizing the booster flyback engines to meet the thrust requirements of the booster element results in more than the required thrust capability on the orbiter element.

The flyback and landing engines on the space shuttle vehicle will be exposed to environmental conditions not normally experienced by conventional aircraft. During the boost and entry phases the engines will be subjected to high vibration loads while they are inoperative. This may cause brinelling of the bearings or other damage. Following the boost phase the engine will be exposed to a weightless, vacuum environment. The orbiter engines will be in this space environment for extended periods of time. The engine lube and fuel systems will probably require redesign. Engine metals and materials will have to be evaluated to assure their compatibility with the space environment. These requirements will have to be considered in the development or modification of the airbreathing engines for the space shuttle vehicle.

4.2 SYNTHESIS PROGRAM PERFORMANCE WEIGHT BASIS

During the space shuttle vehicle study, vehicle element weights have varied over a wide range, aerodynamic parameters have varied with vehicle shape, and different numbers of engines per element have been considered. The procedure used to assure reasonable engine weights and dimensions over the wide vehicle configuration range is described in this section.

The engine performance, weight, and dimensional data were based on data compiled on the General Electric turbofan subsonic cruise TF34 engine for the VS(X) program and described in Volume 10, "Power Plant Analysis" of the VS(X) (S-3A) proposal (Reference 4-1). This engine is presently in development and is scheduled for flight qualification in 1972. Several growth steps have been projected for this engine. The Growth Step III version, which achieves a 36 percent thrust increase by increasing the fan diameter, increasing the turbine inlet temperature, and incorporating a transpiration-type turbine cooling system, was used as the base engine. Characteristics

of this engine were rubberized and defined by scaling equations for use in vehicle synthesis. Characteristics and scaling equations assumed for the engine include:

- TF 34 Growth Step III Characteristics
- Maximum SLS Thrust, lb (F_N*)
- S.L. Specific Fuel Consumption, lb/hr/lb
- Scaling Equations
  - Diameter, in.
  - Length, in.
  - Installed Weight

} See Confidential Addendum

Figure 4-2 shows the engine weight, diameter, and length as functions of maximum sea level static thrust.

Figure 4-2. Engine Scaling Data

The installed performance of the engine relative to the maximum sea level static performance was determined from Reference 4-1. Performance summarized in Reference 4-1 for seven different conditions indicates losses ranging from 5.4 percent to 13.1 percent of the delivered net thrust.

Figures 4-3 and 4-4 show the ratio of available thrust to maximum sea level static thrust and the corresponding SFCs as functions of velocity for the maximum continuous thrust rating condition.

The cruise velocity of the vehicle at a particular condition has been assumed to be the velocity that the vehicle would fly with a maximum L/D. The equation used to determine the cruise velocity is:

$$\text{Vehicle Weight} = \frac{1}{2} \text{ (Air Density) (Velocity)}^2 \text{ (Reference Area) } (C_L)$$

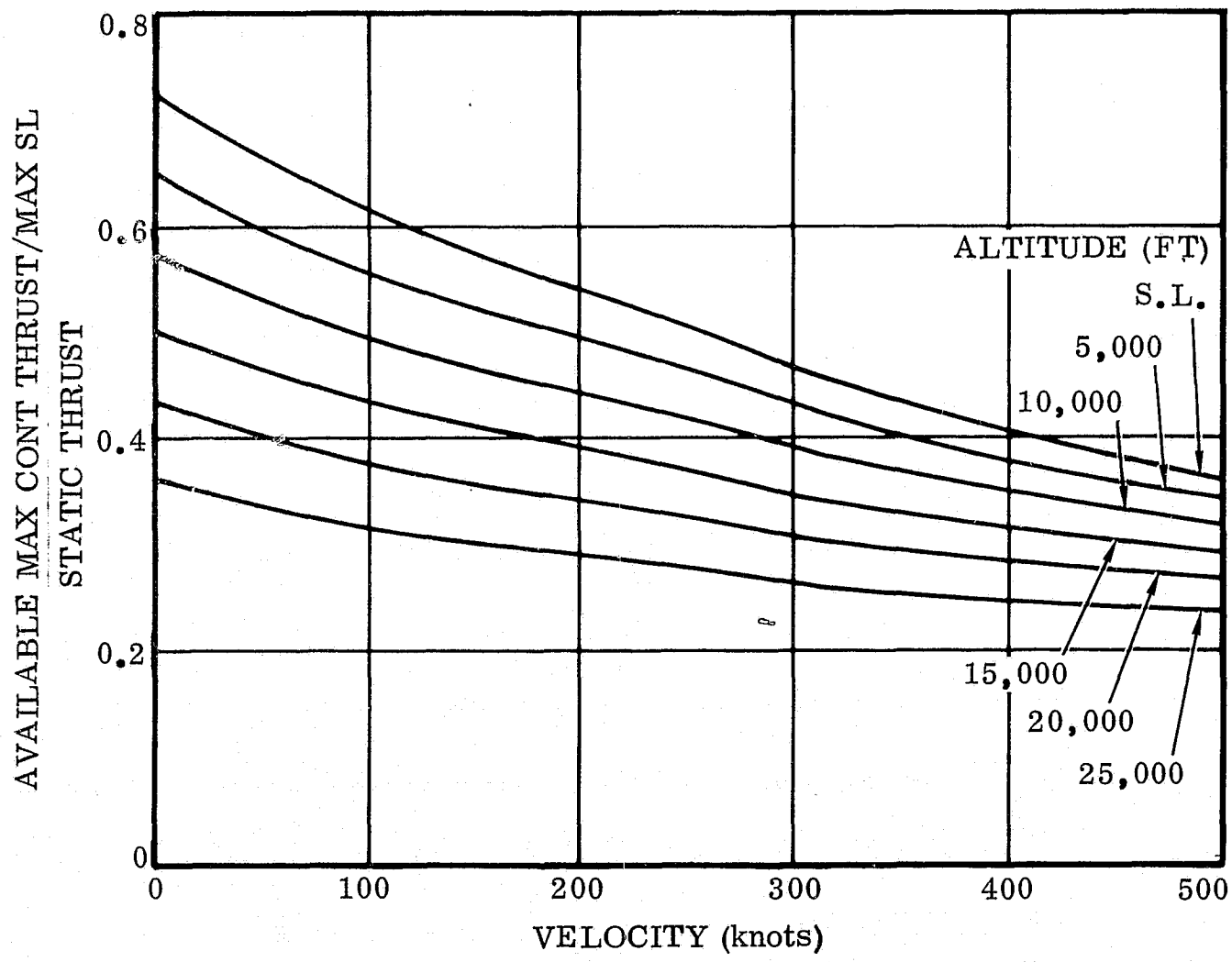


Figure 4-3. Thrust Ratio versus Altitude and Velocity

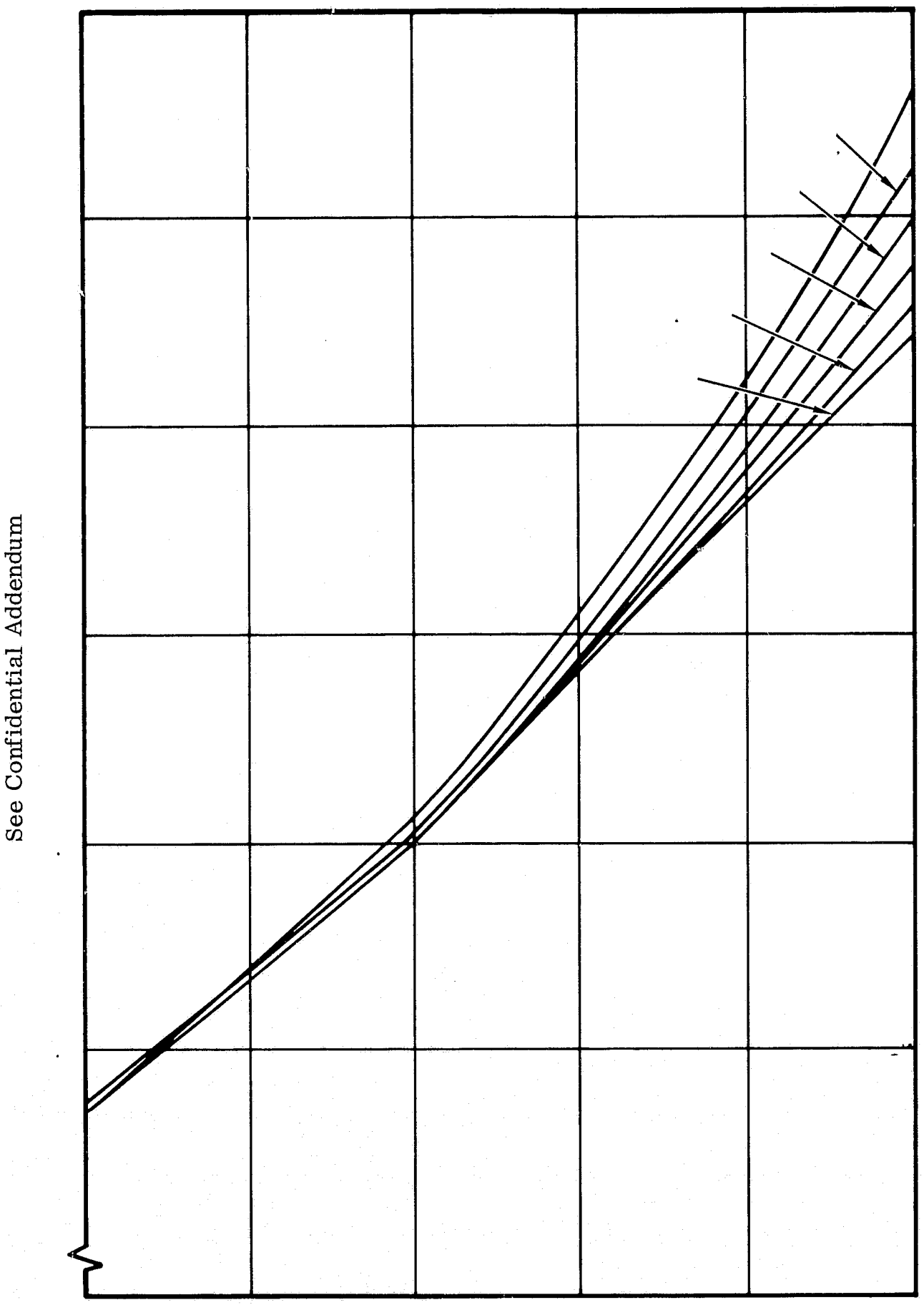


Figure 4-4. Specific Fuel Consumption versus Altitude and Velocity

Delivered thrust required is:

Total Delivered Thrust = Vehicle Weight/L/D

Maximum L/D and C_L at maximum L/D have been estimated from wind tunnel data. Final estimates used to size the flyback engines are:

	FR-4 & FR-3 Orbiter	FR-3 Booster
Max Cruise L/D	7.8	7.2
C_L at Max L/D	0.55	0.45

The airbreathing engine weight equation in the synthesis program is in the form

$$\text{Weight} = (C_1)\ (\text{Vehicle Flyback Weight}) + C_2$$

C_1 (total engine weight/vehicle flyback weight) has been calculated using equations described above. C_2 is the 200-pounds per engine in the engine weight scaling equation. In determining the value of C_1 the best estimate of vehicle flyback weight available in each case was used to determine the required thrust and initially size the engines.

The booster fuel weight was calculated by the synthesis program from the Brequet range equation:

$$\text{Range (n.mi.)} = \frac{V_{(\text{knots})}}{\text{SFC (lb/hr/lb)}} \times \text{L/D} \times \ln \frac{W_{\text{initial}}}{W_{\text{final}}}$$

The range and entry weight were determined from the program trajectory and weight calculations. The velocity, L/D, and SFC were input to the program. Since the booster element must have cruise-back capability with one engine out, the values corresponding to this condition (lower altitude, lower velocity, emergency thrust rating) were input. This resulted in a higher fuel weight than the normal cruise condition would have required.

The orbiter fuel weight is a constant multiplied by the orbiter entry weight. The orbiter fuel has been assumed to be sufficient to cruise for 10 minutes at sea level conditions.

4.3 CANDIDATE ENGINES

To assure reasonable engine weights and dimensions for parametric studies using the synthesis computer program, the airbreathing engines were scaled as describe in the

previous section. A study was performed to determine the effect of different types of airbreathing engines on the airbreathing system weight and to assure that the high bypass turbofan is the correct choice.

The two major contributors to the airbreathing engine system weight are the engine weights and fuel weights. Maximum sea level static thrust (MSLST)/bare engine weight ratios for high-bypass turbofan engines now in development vary between 5.4 and 7.3. The F401 non-afterburning low-bypass (0.8) engine under study for Pratt and Whitney Aircraft has a thrust to weight ratio of 7.0. For the weight comparison study a F/W ratio of 6.4 was assumed for the high bypass fan engine and a F/W ratio of 7.0 was assumed for the turbojet engine. For a booster element weighing approximately 350,000 lb, the installed total engine weights per element, (three engines per element) assuming the same installation weights for either engine type, are:

Turbofan	22,140 lb
Turbojet	21,255 lb

The specific fuel consumption decreases with bypass ratio. Figure 4-5 shows the trend in SFC for a turbojet, a low-bypass fanjet (0.8), and a high-bypass fanjet (6) at a 15,000 ft cruise condition.

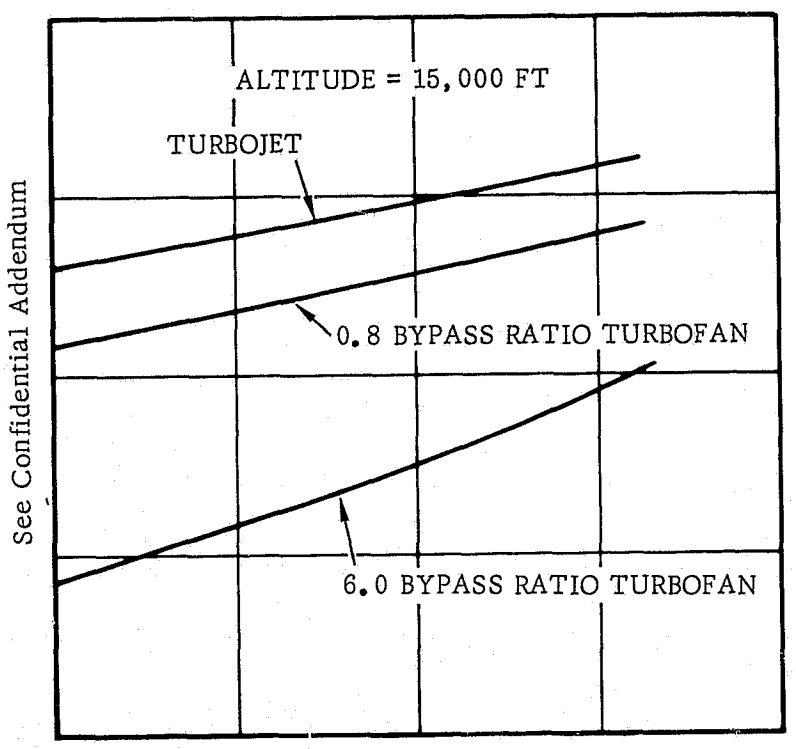


Figure 4-5. Effect of Bypass Ratio and Velocity on Specific Fuel Consumption

The booster elements cruise back approximately 250 n.mi. at a velocity of 230 knots. The fuel weights required by the two types of engines are:

Turbofan	24, 800 lb
Turbojet	37, 100 lb

Figure 4-6 shows the comparative weights of the two systems. The turbojet engines are slightly lighter; however, the turbojet fuel plus engine weight is 24 percent heavier than the turbofan system. Use of hydrogen fuel instead of JP-4 would reduce the total weight difference between turbojet and turbofan to 11 percent.

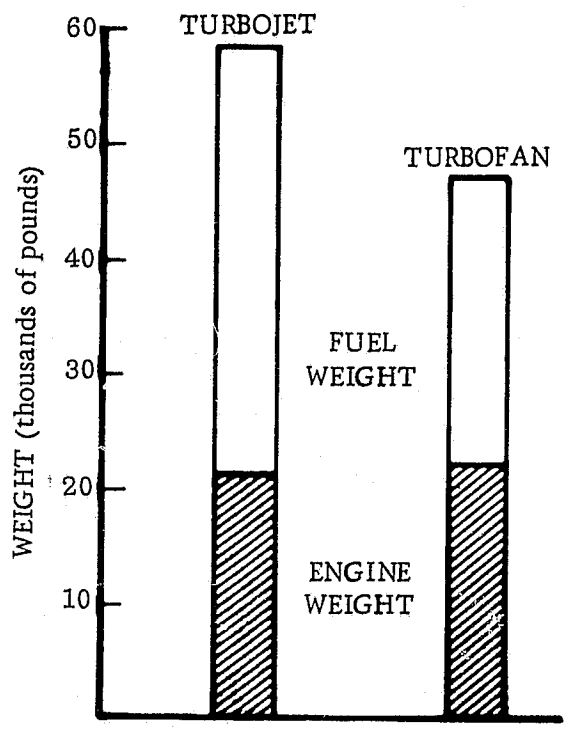


Figure 4-6. Booster Element Airbreathing Engine System Weight Comparison

The turbojet engine with afterburner has been considered and is an approach that will reduce the engine weight since the thrust of an engine can be increased by 50 percent with an increase in weight of approximately 15 percent. However, specific fuel consumption increases by about 150 percent. The resulting weight for a system sized for an afterburning turbojet would be:

Engine Weight	16, 300 lb
Fuel Weight	90, 000 lb
Total	106, 300 lb

compared to 46, 926 lb for the turbofan system and 58, 355 lb for the non-afterburning turbojet. The high thrust-to-weight ratio and low fuel consumption of the advanced high-bypass turbofan engine best meet the requirements of the space shuttle boosters.

Development of a new engine to match the requirements of the space shuttle vehicle would be so expensive that the limited number of engines required for the program makes it almost prohibitive. It is highly desirable, therefore, to use an engine that is developed for another program, making only the modifications required to adapt it to the special environment of the space shuttle vehicle.

Engines recently developed or presently under development are shown in Table 4-1.

Table 4-1. Candidate Airbreathing Engines

Engine	TF 34-2	RB 211-22	CF6-6	TF39-1	JT9D-15
Manufacturer	GE	Rolls Royce	GE	GE	P&W
Application	S3A	L-1011	DC-10	C-5	DC-10
Maximum SLS Thrust, lb SFC @ MSLST, lb/hr/lb Diameter, in. Length, in.* Weight, lb*	See Confidential Addendum				

*Bare engine without tailpipe and inlet duct.

Between 10,000 pounds and 40,000 pounds there are presently no advanced turbofan engines that will be available. There are several older engines with MSLST in the 10,000 pounds to 21,000 pounds range. These engines however have thrust-to-weight ratios in the 4 to 4.5 range and SFC ranges of 0.55 to 0.65 lb/hr/lb. Because of the 50 percent or greater increase in fuel consumption and lower thrust-to-weight ratios of these older engines, using them would result in heavier airbreathing system weights for the booster element which must fly back several hundred miles, even for vehicle weights in the 150,000 to 300,000-pound range that require greater than five of the 9280-pound-thrust engines and where three of the 40,000-pound engines would deliver more thrust than required. Of the engines available, therefore, the advanced turbofan engines listed in Table 4-1 are the best candidates. Growth versions of these engines that may be available within the next 5 to 10 years have also been considered. They include:

TF34 Growth Step III
RB 211-43
RB 211-56
TF 39-F6
CF 6-50A

} See Confidential Addendum

The orbiter element is required to have the capability to make a powered approach and landing at sea level. The operating time of the engines will be 10 minutes or less and the engine to deliver the required thrust at sea level can be much smaller than the booster engine, which must operate at 15,000 feet. If other requirements such as ferry capability, common engines on the orbiter and booster, and power takeoff are not imposed, a high thrust-to-weight engine developed for usage as a lift engine or short-duration thrust assist would provide a significant weight saving in the orbiter.

The XJ99 jet engine under development by Rolls-Royce Limited and Allison Division of General Motors Corporation is a candidate engine of this type. Demonstrator engines have been built and tested. Parameters of the engine include:

Main Jet Thrust SFC, lb/hr/lb Basic Dry Weight, lb Length, in. Diameter, in.	See Confidential Addendum

Scaling information indicates a recommended thrust range of 4750 pounds to 14,400 pounds.

4.4 INSTALLATION

A study was performed to determine the effect on the airbreathing system weight and vehicle launch weight of the number of engines per element for the FR-1 vehicle. A booster entry weight of 320,000 pounds and scaled engines from data described in Section 4.2 were used in the study. Two engines per element, sized to provide cruise capability at 15,000 feet, will not provide sufficient thrust for the element to fly at sea level with an engine inoperative so this option was not considered. The effects of three engines per element and four engines per element were compared.

The scaling law for engine weight results in an increase in total engine weight per element of 1.8 percent for the larger engines in the three-engine configuration. The four-engine configuration also has the capability, with one engine out, to fly at 7700 feet altitude at a velocity of 184 knots compared to 3200 feet and 173 knots for the three-engine configuration. This provides a fuel saving of 5.2 percent for a 240-n. mi. flight. The resulting launch weight saving is 0.5 percent. From a weight and engine-out capability standpoint only, the four-engine configuration is slightly better. The weight penalty is small and there are other factors which favor the three-engine configuration. They include:

a. Three engines, when retracted, fit within the vehicle outline.

b. The engines deploy on a simple single-axis pivot located close to a major structural bulkhead to provide short load paths for thrust and dynamic loads.

c. When extended, the engines are widely separated and isolated from the vehicle, minimizing interaction stability problems and fire hazards.

d. Nacelles, when extended, provide good access for maintenance (with proper work platforms).

The three-engine configuration was chosen for installation studies. Figure 4-7 shows the flyback engine installation in the FR-1 vehicle. The engines in the figure are scaled engines having a MSLST per engine of 37,400 lb. Unconventional aspects of the engine installation include:

a. Stowage and deployment concept.

b. Inverting the upper engine while stowed.

c. Exposure of the engine to possible reverse rotation during deployment.

The stowage and deployment method, while unconventional for engine installations, should present no major problems. Technology developed in the deployment of landing gear systems and use of variable sweep wings in recent years should provide the answers to design of the mounting, pivoting, and actuating systems of the deployable pylon.

Inversion of the engine during stowage will require that the engine oil system be designed to prevent loss of oil and undesirable trapping of air. In any case, the design of the engine fluid systems to meet the requirements for stowage in the zero-g and vacuum environment of space and the accelerations in various directions during mission trajectories should provide the solution of the problem.

During the initial period of the deployment cycle, the engine will be subjected to airflow up the tailpipe and fan exit. This will tend to spin the engine rotors in the reverse direction. The reverse airflow will exist for about one-half of the deployment movement, then the nacelle nose will begin to face into the wind and windmilling in the normal direction will start. Reverse windmilling during deployment will cause no problem if time is no more than 10-15 seconds (Reference 4-2). This requires that the portion of deployment time during which reverse windmilling occurs be no longer than 15 seconds. A total deployment time of 20-30 seconds is feasible.

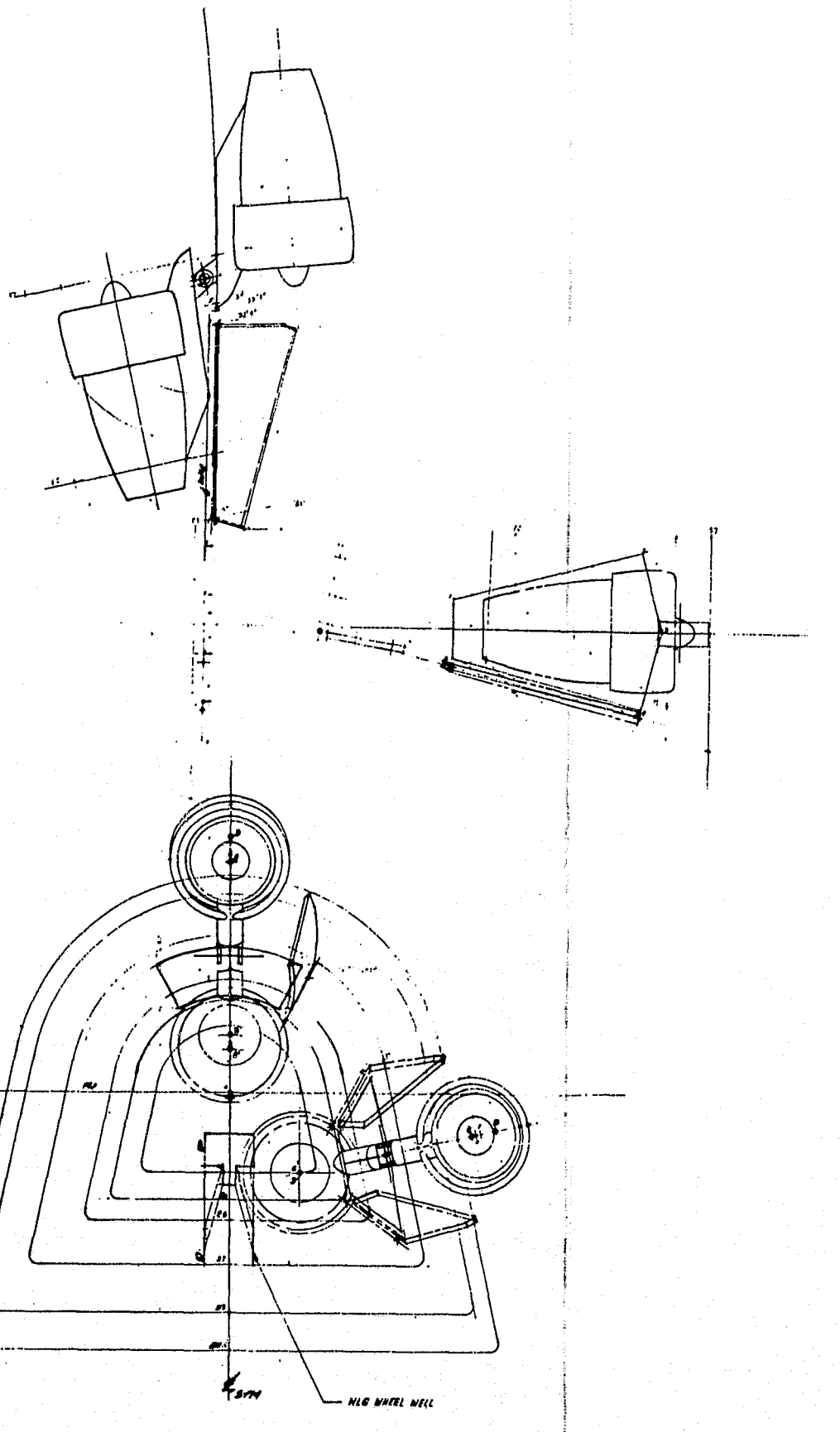
Two alternate engine installations were drawn to see how engines currently in development will affect the FR-1 vehicle.

Figure 4-8 shows the Rolls Royce RB211 engine and Figure 4-9 shows the General Electric CF6 engine. Both of these engines are larger than the scaled engine assumed for the basic installation shown in Figure 4-7. The following modifications to the FR-1 vehicle are indicated:

a. The bulkhead at Station 21 will have to be moved forward.

b. The fuselage doors will have to be enlarged.

c. The fuselage lines should be expanded slightly in the engine compartment area to allow more clearance over the basic structure.



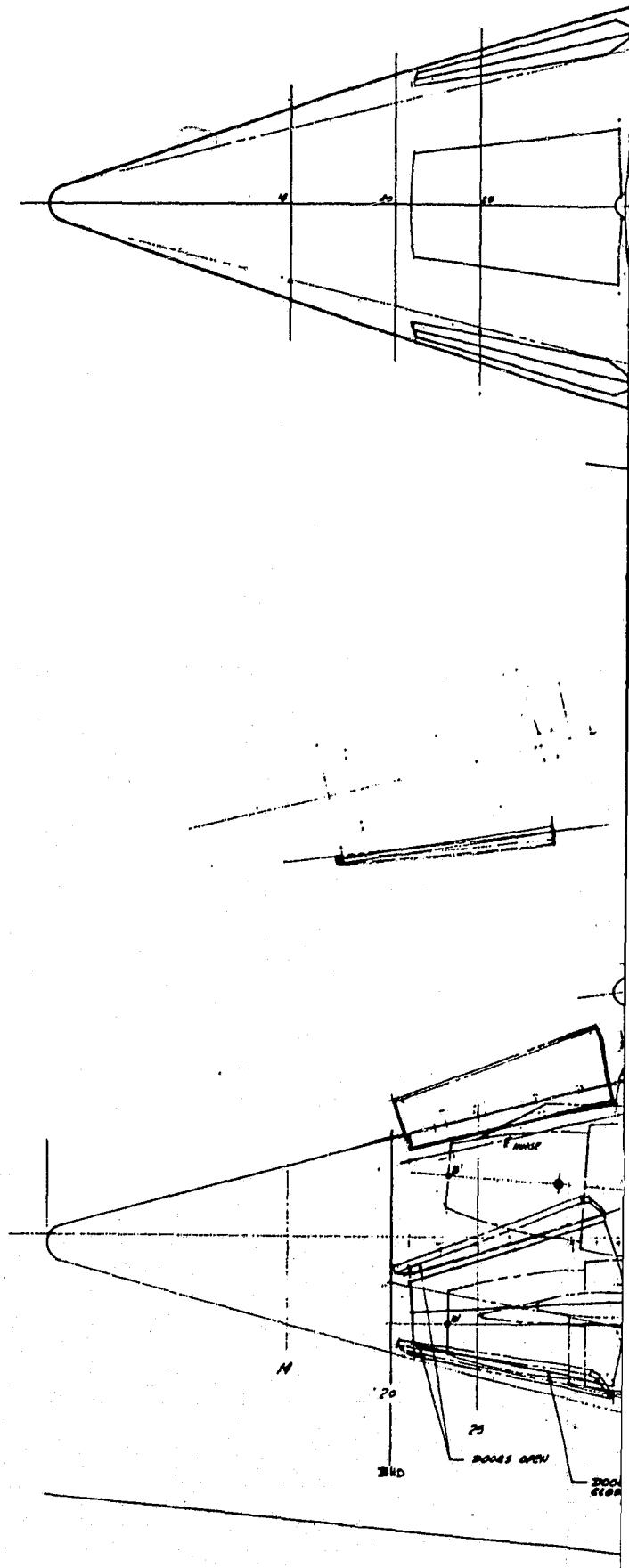
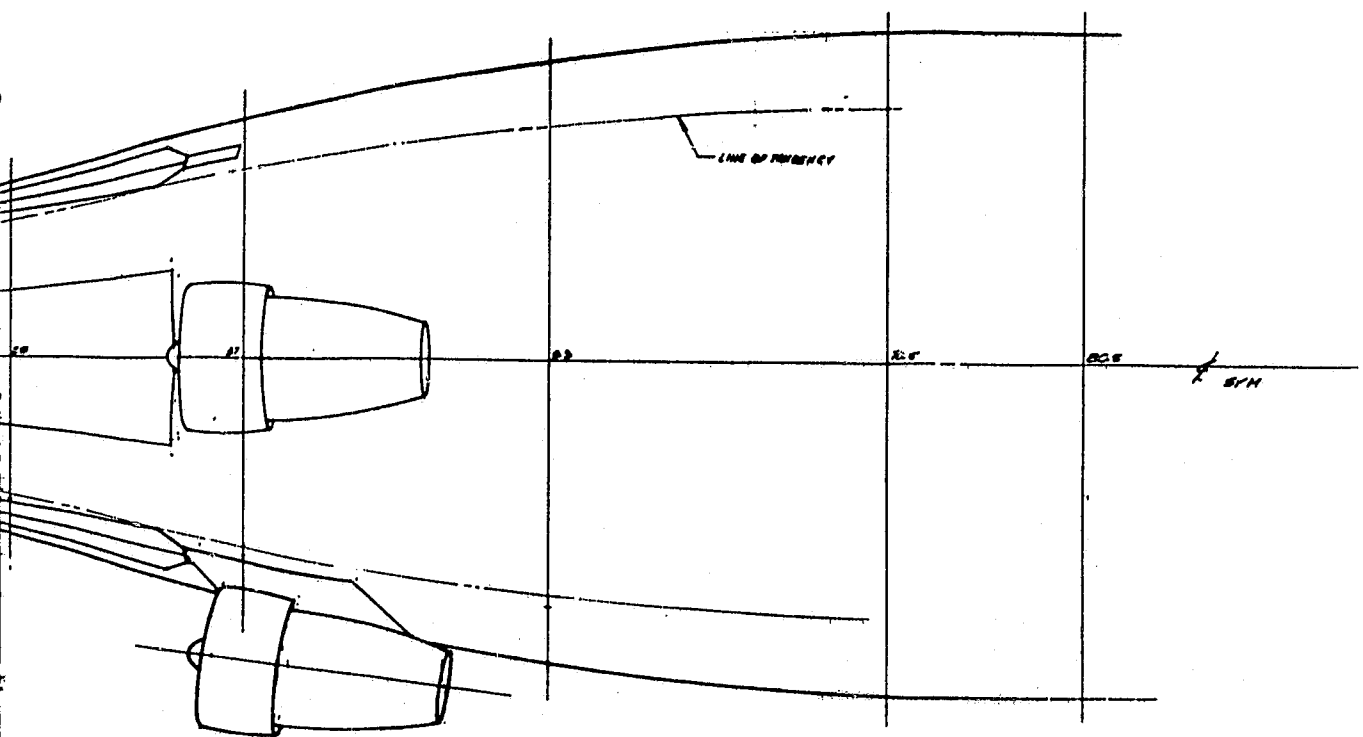
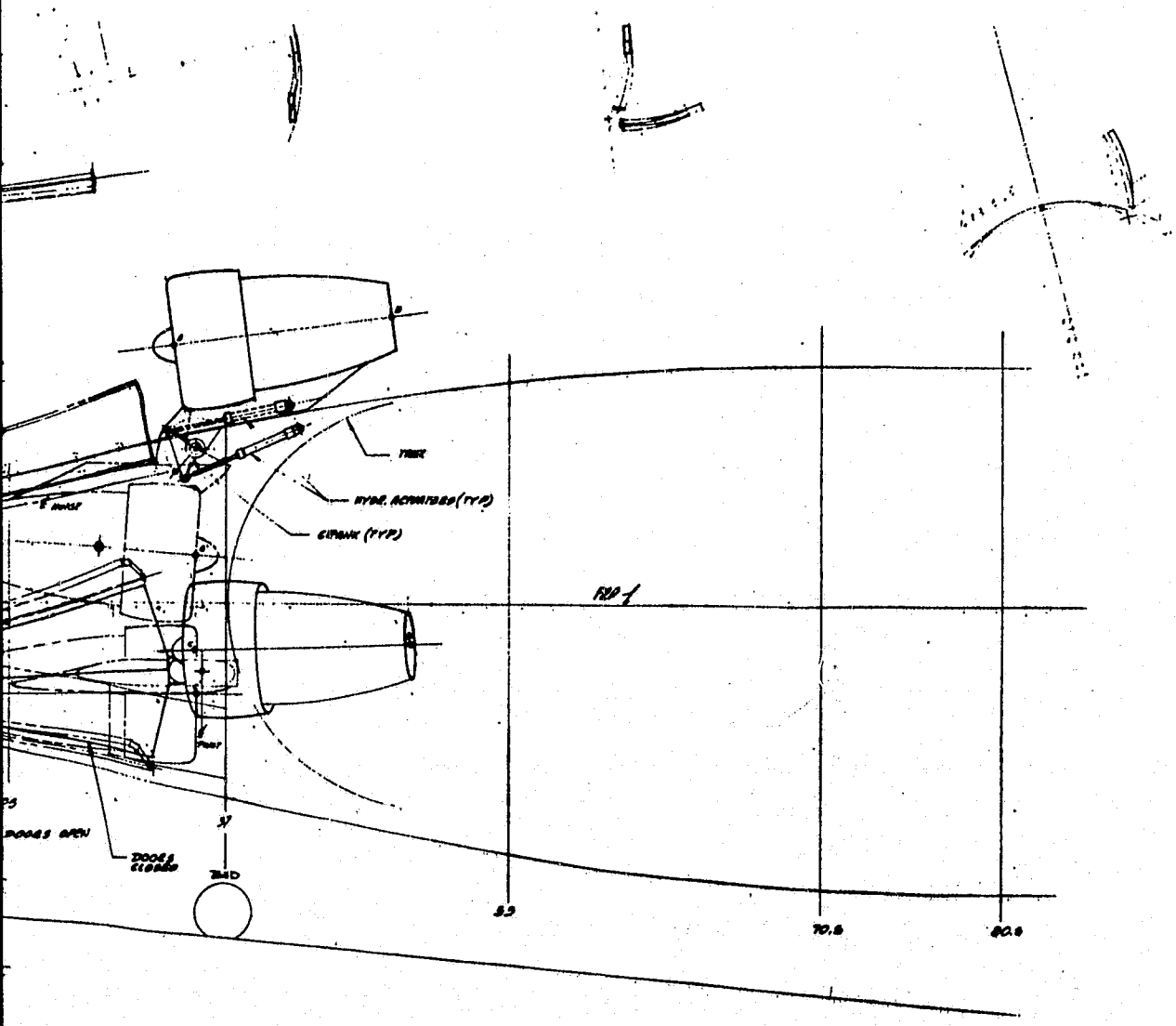


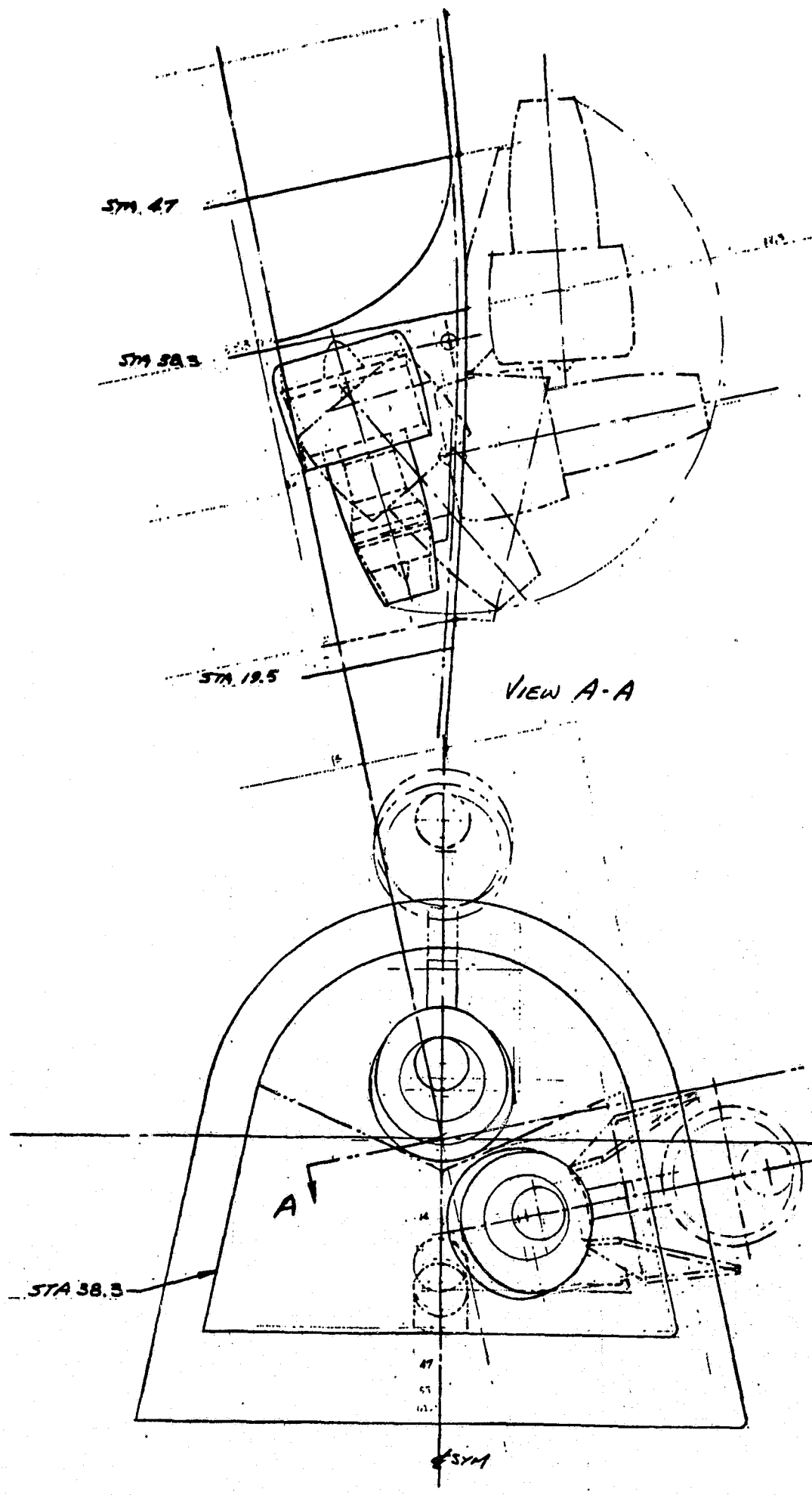
Figure 4-7. Flyback Engine Arrangement, Three-Engine Version, Scaled Engines

FOLDOUT FRAME 1

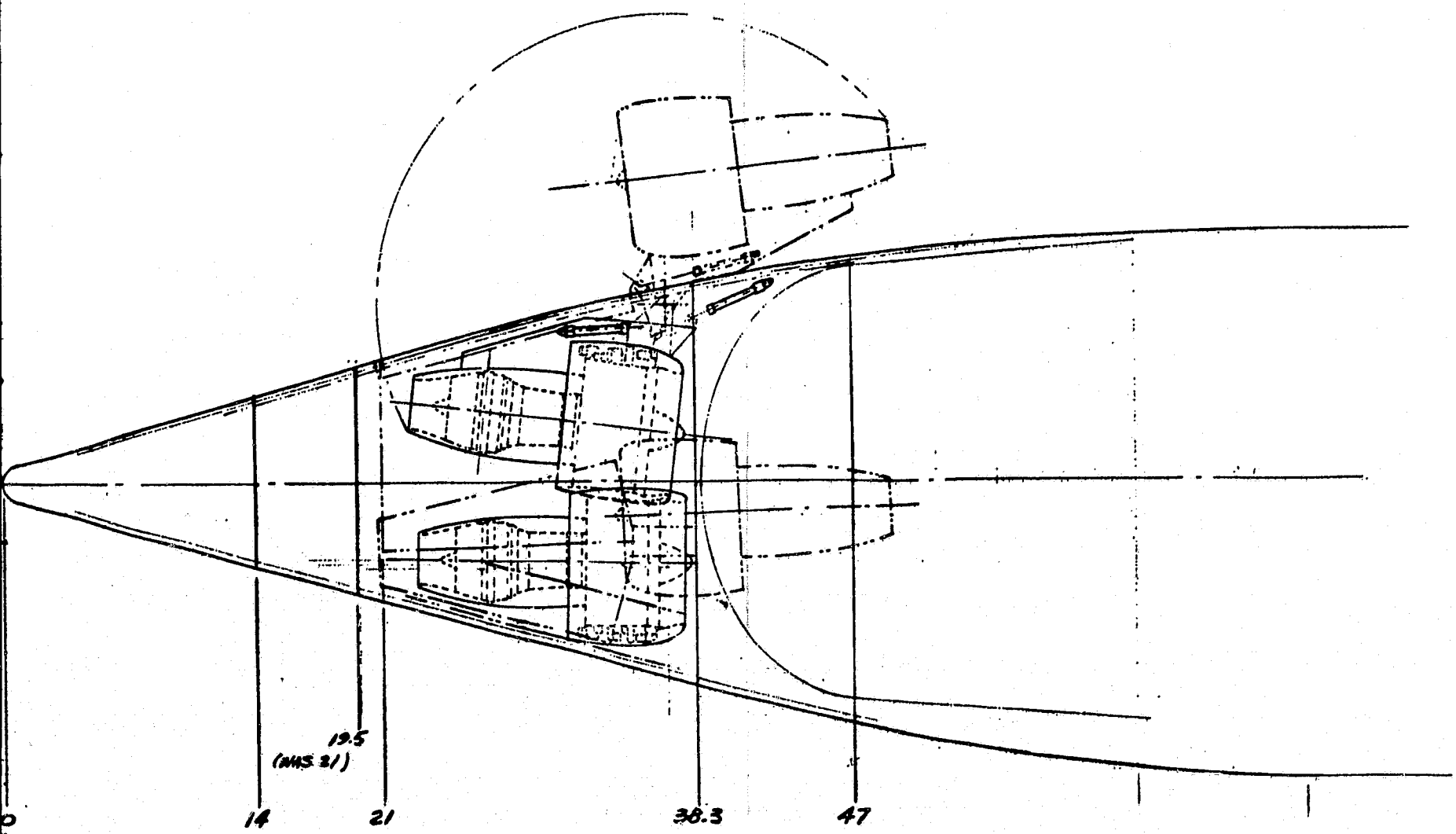




FOLDOUT FRAME 2



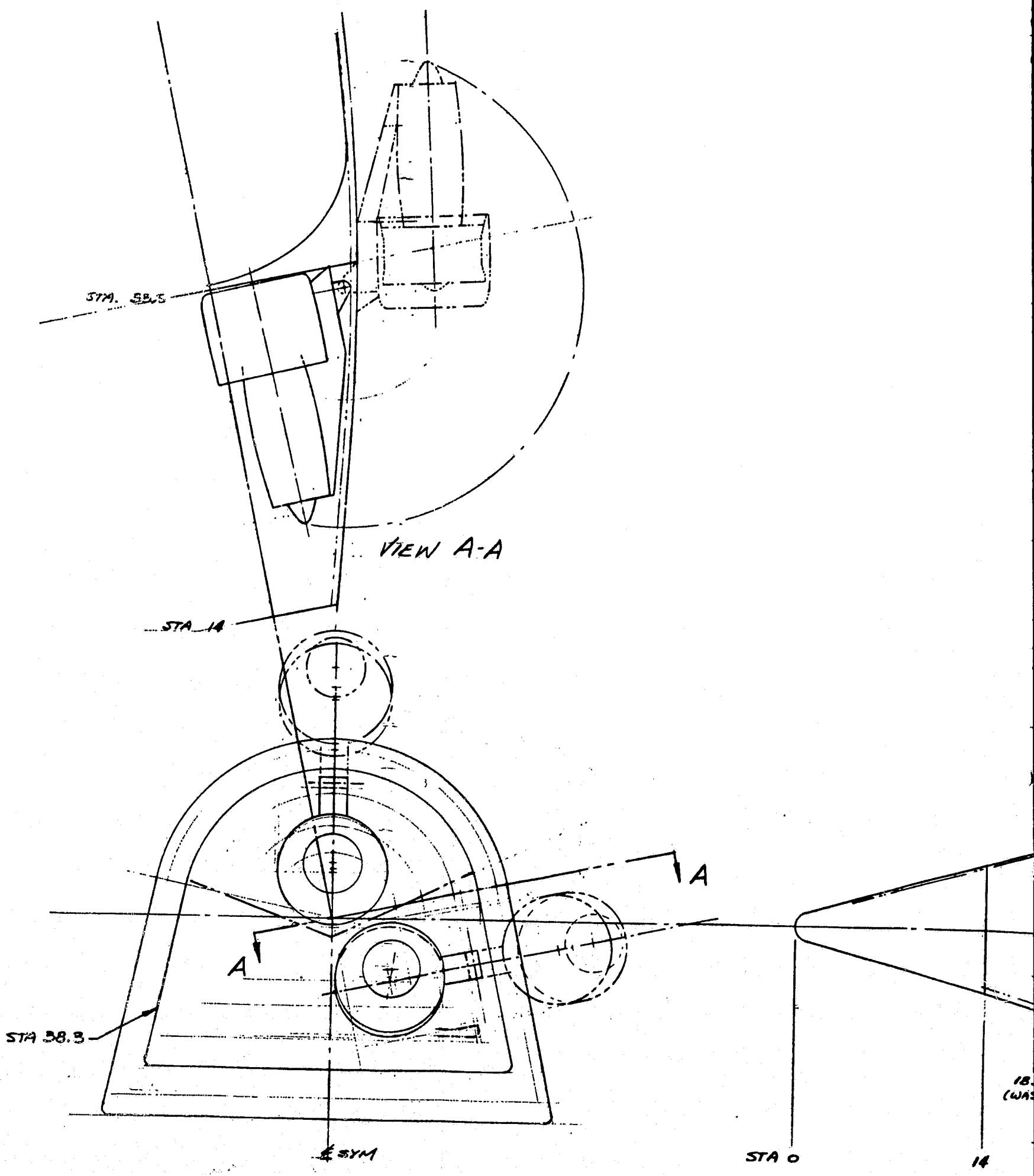
FOLDOUT FRAME 1



SIDE VIEW

Figure 4-8. Flyback Engine Arrangement, Three RB-211 Rolls Royce Engines

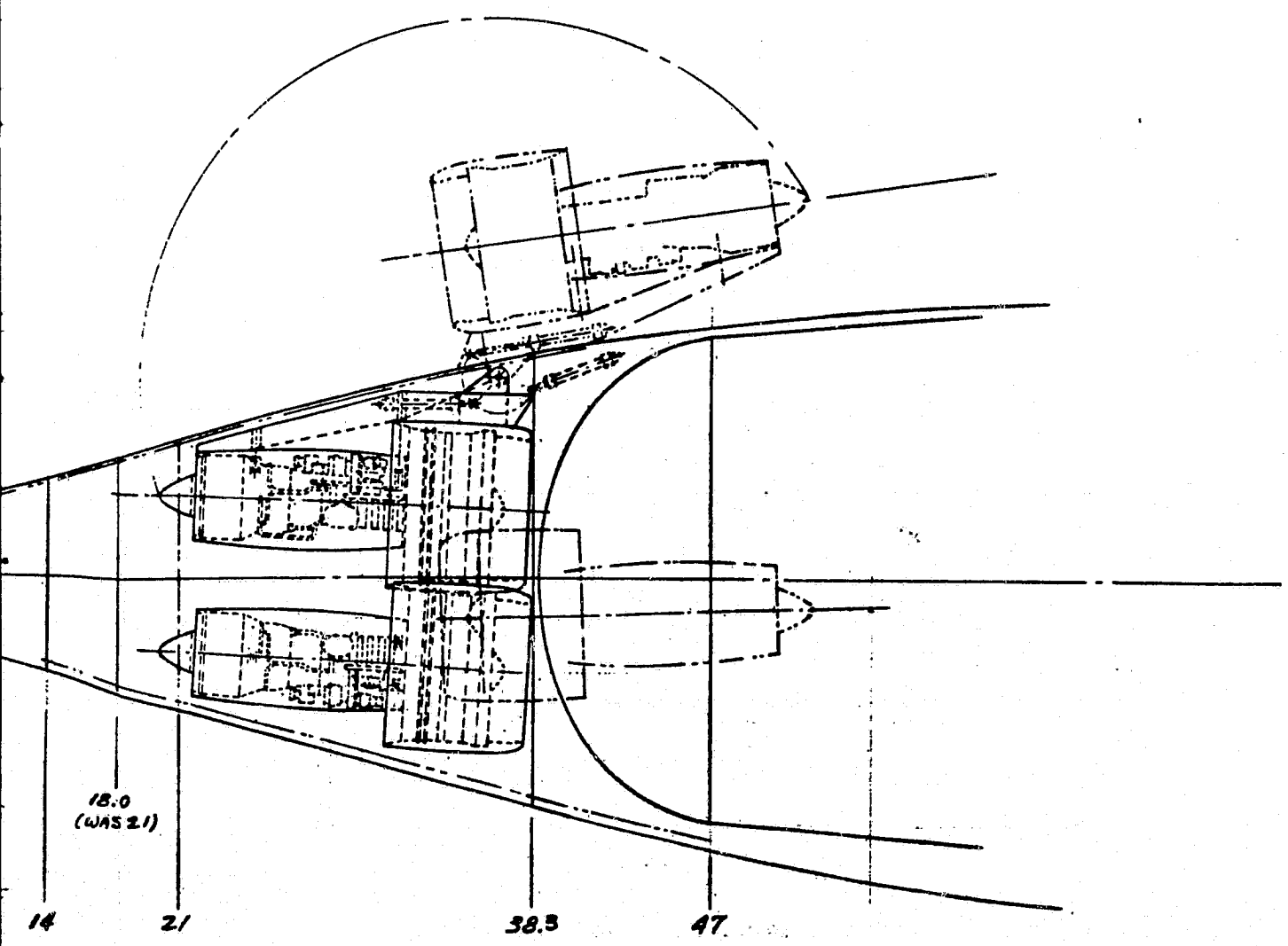
FOLDOUT FRAME 2



VIEW LOOKING AFT

Figure 4-9. Flyback Engine Arrangement, Three CF6 (Modified) GE Engines

FOLDOUT FRAME 1



SIDE VIEW

FOLDOUT FRAME 2

The three-engine configuration has certain installation advantages as indicated above. The four-engine configuration has a slight weight advantage. The number of engines chosen for a particular vehicle will be based on the best match that can be made between available engines and the required thrust of the vehicle.

Use of high thrust-to-weight, short-duration engines such as the XJ99 has been considered for the orbiter element. These engines are much smaller than the turbofan cruise engines so the installation and deployment problem should be minimized. An alternate installation that has been considered and appears feasible is to mount these small engines on the wings so that they are deployed when the wings are extended. Figure 4-10 shows this installation.

4.5 EFFECT OF HYDROGEN FUEL ON FR-3 WEIGHTS

Hydrogen has been considered as a replacement for JP-4 for the airbreathing flyback engines. Hydrogen has a heat content 2.77 times that of JP-4, and residuals remaining in the main propulsion fuel system that would otherwise be wasted may be utilized. However, the density of hydrogen is only about 9 percent of the density of JP-4 so hydrogen would require more volume. The effect of using hydrogen as the airbreathing engine fuel on the weight of the 15-foot-diameter payload FR-3 vehicle, as mentioned in Section 2.4.3, has therefore been studied.

The booster element has 46,900 pounds of JP-4 fuel; the orbiter has 2868 pounds. Weight and volume changes associated with the use of hydrogen fuel were determined, and their effect on the launch weight of the FR-3 vehicle was calculated using weight and volume sensitivity factors. It was assumed that main-propulsion residual hydrogen in the booster could be used by the airbreathing engines but that a special fuel feed system (compressor, etc.) would have to be added. The main tank was enlarged to provide for the additional H_2 required. In the orbiter, it was assumed that main-propulsion residual hydrogen has been used by the ACPS but that the ACPS fuel feed subsystem could also be used for the airbreathing systems. Main fuel tanks were again enlarged for the additional H_2. Table 4-2 summarizes the weight and volume effects of using hydrogen as fuel in the airbreathing system. Results of the study indicate a potential launch weight decrease of 218,000 pounds, which is 5 percent of the total launch weight.

4.6 CONCLUSIONS

The space shuttle vehicle has airbreathing engines on the booster element which provide it with the capability to fly back several hundred miles and land at the landing site. The orbiter element airbreathing engines provide it with the capability to make a powered approach and landing. Conclusions made as a result of the study include:

a. Advanced high-bypass turbofan engines give the lightest flyback and landing subsystem on the booster.

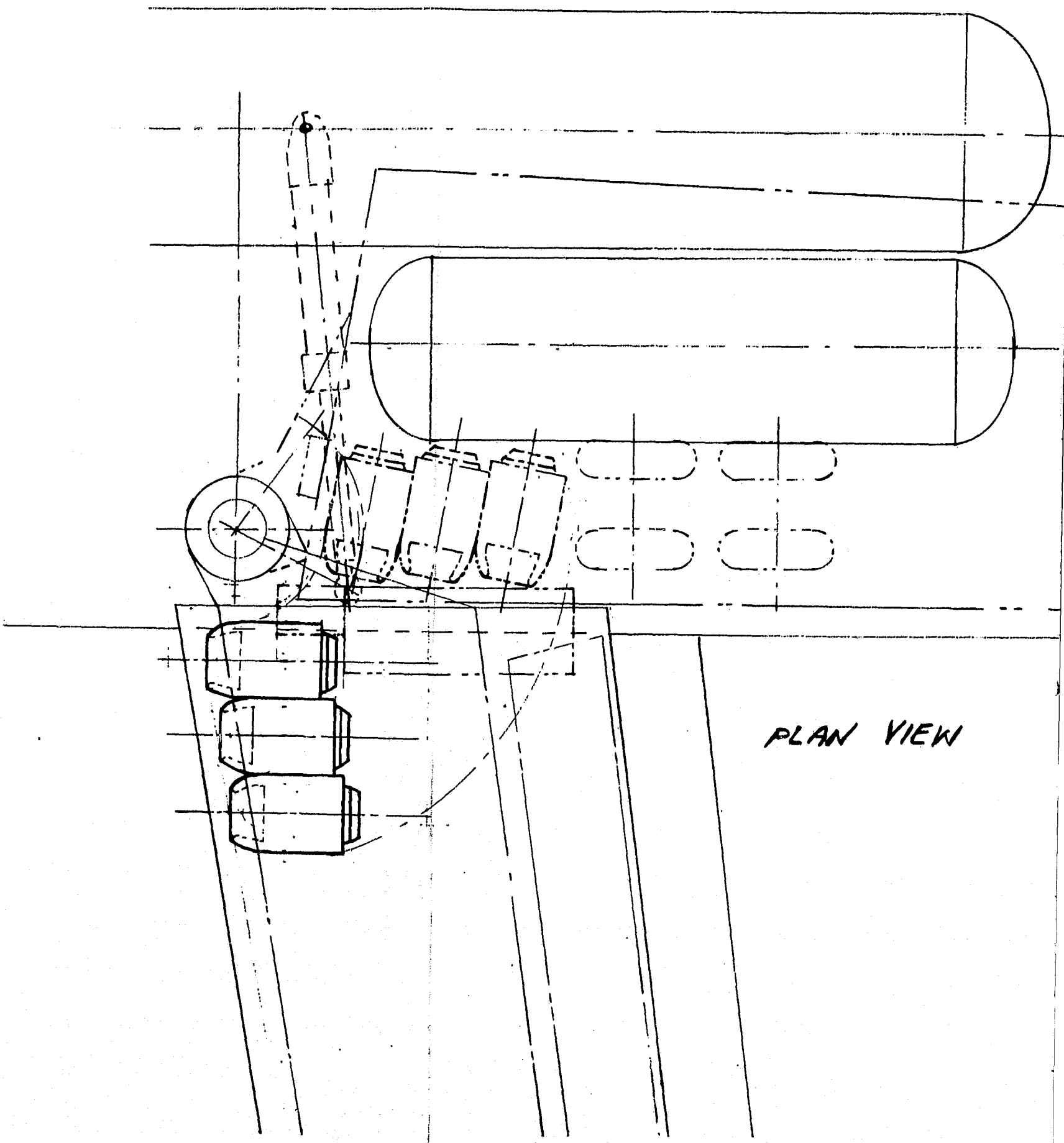
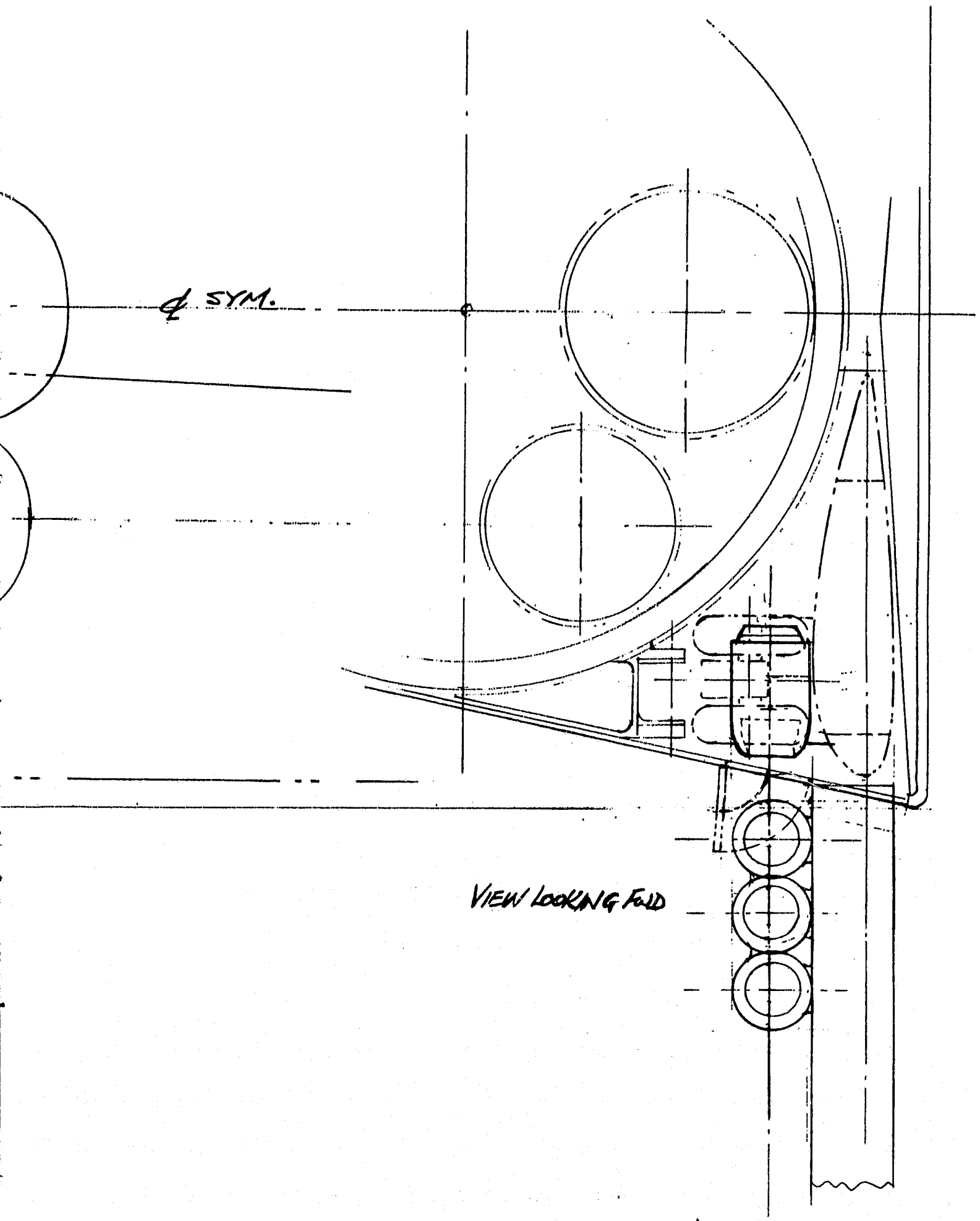


Figure 4-10. Lift Engine Installation, Six Engines

FOLDOUT FRAME 1



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SCALE 1/10 RAVENSCROFT 9-25-69

FOLDOUT FRAME 2

Table 4-2. Comparison of Using JP-4 or H_2 Fuel in Airbreathing Engines for the Final 15-Foot-Diameter Payload FR-3 Vehicle

Booster	
JP-4 Fuel Weight, lb	46,900
JP-4 Tank Weight, lb	4,740
H_2 Fuel Weight, lb	16,930
H_2 Total Residual, lb	5,547
H_2 Usable Residual, lb	3,300
$H_2 \Delta$Tank Weight, lb	1,739
H_2 Fuel Feed System Weight, lb	500
JP-4 Volume, ft^3	963
H_2 Volume, ft^3	3,077
Orbiter	
JP-4 Fuel Weight, lb	2,868
JP-4 Tank Weight, lb	337
H_2 Fuel Weight, lb	1,035
$H_2 \Delta$Tank Weight, lb	132
JP-4 Volume, ft^3	59
H_2 Volume, ft^3	234
Summary	
Booster Weight Decrease, lb	35,771
Booster Volume Increase, ft^3	2,114
Orbiter Weight Decrease, lb	2,038
Orbiter Volume Increase, ft^3	175
Launch Weight Decrease, lb	218,058
Percent Decrease	5.0

b. A three-engine or greater configuration sized for the booster to cruise at 15,000-feet altitude has the capability to fly back at a lower altitude and velocity with one engine inoperative.

c. The four-engine configuration is slightly lighter than the three-engine configuration, but the three-engine configuration has installation advantages.

d. Development of a new engine to match the requirements of the space shuttle vehicle may make it prohibitive to develop a new engine. An existing engine that best matches the vehicle requirements should be chosen. Modification will be required to adapt this engine to the special environment (high vibration, zero-g, vacuum) of the space shuttle vehicle.

e. The use of short-life, high thrust-to-weight engines developed as lift engines for VTOL aircraft provide a potential weight savings for the orbiter element which is required to have only approach and landing capability.

f. Using hydrogen rather than JP-4 as the flyback subsystem fuel provides a potential launch weight decrease in the order of 5 percent.

g. Best candidate engines for the space shuttle vehicle are the advanced high-bypass turbofan engines with maximum sea level static thrust levels in the 40,000 to 45,000-pound range being developed for subsonic cruise airplanes. These include:

General Electric	CF6-6 and TF39-1
Rolls Royce	RB211-22
Pratt and Whitney	JT9D-15

Growth versions of these engines are also projected.

4.7 REFERENCES

4-1 "Advanced Carrier-Based ASW Air Weapon System VS(X)," Volume 10, Power Plant Analysis, Report GDC-DFM 68-001, Convair division of General Dynamics.

4-2 Letters from General Electric Company, H.E. Garside, Flight Propulsion Representative; Subject: Convair Reusable Space Shuttle, 24 September 1969.

4-3 Performance Specification and Installation Notes - RB 211-22, PP213, July 1968, Rolls Royce Limited.

4-4 Performance Specification - RB211-56, PP217 July 1969, Rolls Royce Limited.